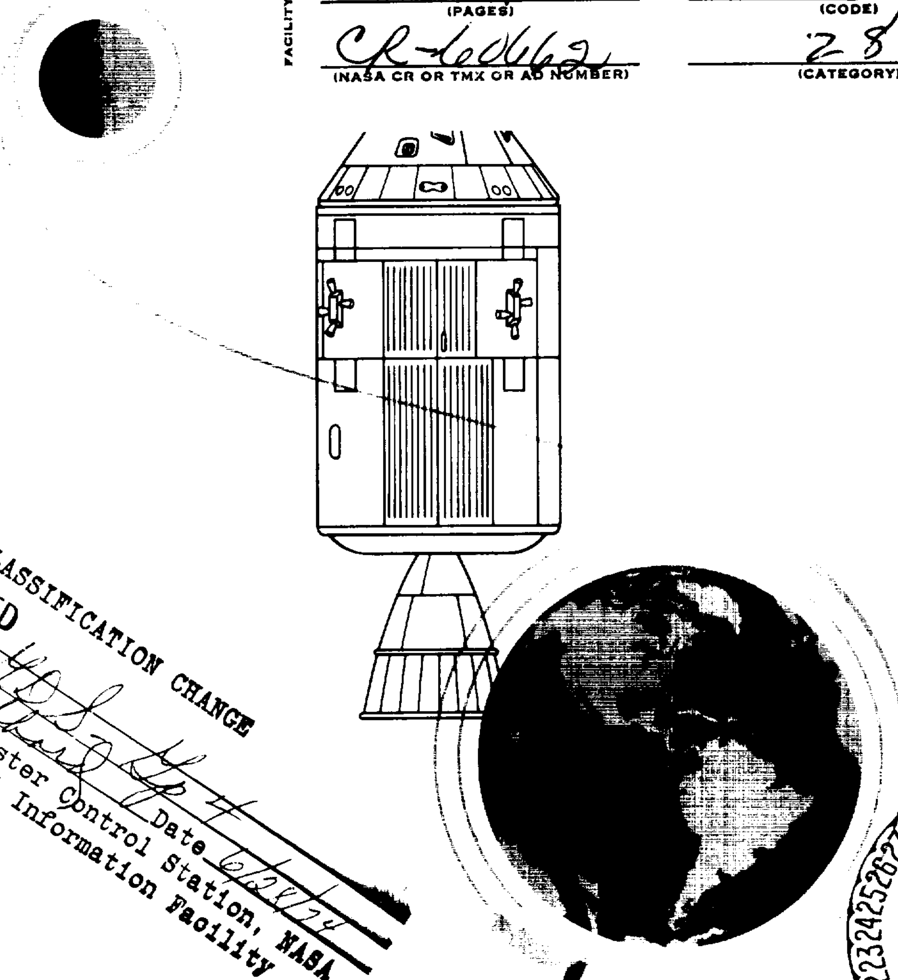
[redacted]

SID 64-1860-12

THE Marquardt CORPORATION

FACILITY FORM 602

X65 12204 (ACCESSION NUMBER)	— (THRU)
178 (PAGES)	20 (CODE)
CR-60662 (NASA CR OR TMX OR AD NUMBER)	28 (CATEGORY)



CLASSIFICATION CHANGE

To UNCLASSIFIED

By authority of NASA SP-4

Changed by [illegible] Date 6/20/74

Classified Document Master Control Station, NASA

Scientific and Technical Information Facility

RECEIVED NASA STI FACILITY INPUT BRANCH JAN 1975

Apollo X Final Report

EXTENDED APOLLO SYSTEMS UTILIZATION STUDY (U)

VOLUME 12 • SERVICE MODULE RCS ENGINE

"Available to U.S. Government Agencies and U. S. Government Contractors Only"

NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION

(NASA-CR-60662) EXTENDED APOLLO SYSTEMS UTILIZATION STUDY. VOLUME 12: SERVICE MODULE RCS ENGINE Final Report (Marquardt Corp.) 178 p

N75-72114

Unclas

[redacted] by law.

[redacted]

1

COPY NUMBER

[redacted]

THE Marquardt CORPORATION VAN NUYS, CALIFORNIA

Prepared for
Manned Spacecraft Center
National Aeronautics and Space Administration
Houston, Texas

Contract No. NAS9-3140

Copy No.

Subcontractor Report No. A1044

SID 64-1860-12

EXTENDED APOLLO SYSTEMS

UTILIZATION STUDY

Final Report

Volume 12

SERVICE MODULE REACTION CONTROL ROCKET ENGINE

16 November 1964

Prepared By

THE MARQUARDT CORPORATION

for A. Belsley
Project Engineer

J. L. Roberts
Program Manager

[redacted]ted [redacted] person is prohibited by law. [redacted]

[redacted]

[redacted] vailable to U.S. Government Agencies and U.S. Government Contractors Only"

[redacted]

THE Marquardt CORPORATION VAN NUYS, CALIFORNIA

Prepared for
Manned Spacecraft Center
National Aeronautics and Space Administration
Houston, Texas

Contract No. NAS9-3140

Copy No.

Subcontractor Report No. A1044

SID 64-1860-12

EXTENDED APOLLO SYSTEMS

UTILIZATION STUDY

Final Report

Volume 12

SERVICE MODULE REACTION CONTROL ROCKET ENGINE

16 November 1964

Prepared By

THE MARQUARDT CORPORATION

for A. Belsley
Project Engineer

J L Roberts
J. L. Roberts
Program Manager

[redacted]

[redacted]

[redacted] Available to U.S. Government Agencies and U.S. Government Contractors Only

FOREWORD

This volume constitutes a portion of the Apollo X Study final report (SID 64-1860) prepared as part of the Extended Apollo Systems Utilization Study conducted by the Space and Information Systems Division of North American Aviation, Inc., for the National Aeronautics and Space Administration's Manned Spacecraft Center under Contract NAS9-3140, dated 6 July 1964. S&ID acknowledges the outstanding technical contributions made to the study by a number of companies; these organizations are identified below along with the title of the report for which they are responsible.

The final report has been prepared in the series of 23 volumes listed below:

1. Summary
2. Mission and Performance Analysis
3. Experimental Programs
4. Configurations, Structures, and Weights
5. Mission Plans and Functions
6. Environmental Control System (AiResearch)
7. Fuel Cells (Pratt & Whitney)
8. Alternate Power Source (General Electric)
9. Cryogenic Storage System (Beech Aircraft)
10. Service Module SPS Engine (Aerojet)
11. Command Module RCS Engine (Rocketdyne)
12. Service Module RCS Engine (Marquardt)
13. RCS Propellant Tanks (Bell)
14. Guidance and Navigation (AC Spark plug)
15. Alternate Guidance System (Autonetics)
16. Guidance Computer (Raytheon)
17. Stabilization and Control System (Honeywell)
18. Communications and Data System (Collins Radio)
19. Earth Landing System (Northrop Ventura)
20. Subsystems Supplement
21. Reliability and GSE
22. Development Planning
23. Condensed Summary

PREFACE

The NASA Manned Spacecraft Center has examined for the past several years the application of the Apollo spacecraft to missions alternate to the basic lunar landing mission for which it is currently designed. The overall objective of these studies is an assessment of the advantages and disadvantages associated with the application of developed hardware to these other potential missions.

NASA studies of near-term applications, which represent possible initial extensions to current Apollo capabilities, include use of the Apollo spacecraft as (1) a space station logistics resupply vehicle (carrying up to six men), (2) an earth-orbital space station or experimental laboratory, (3) a lunar mapping and survey vehicle, etc. This report presents the results of the most recently concluded studies of the Extended-Mission Apollo, in which the application to a 45-day earth orbital laboratory role and to an extended lunar orbit mission have been examined in depth.

The initial Extended-Mission Apollo study, initiated in August 1963 under contract to NASA/MSC, examined the suitability of Apollo as an earth-orbital biomedical/behavioral experimental laboratory. These experiments were to provide a basis from which man's suitability for protracted space missions could be determined. Three basic configuration concepts, indicated in Figure 1, were investigated throughout the study. Configuration Concept I utilizes only the Apollo command and service modules (CSM), with experimental in-orbit work space made available in the command module by the elimination of one crewman from the current crew size of three. The Apollo CSM subsystems were modified to sustain orbital operations for periods of up to 120 days without resupply. Configuration Concept II consisted of the Apollo CSM plus a 5600 cubic foot laboratory module built within the geometric limits of the LEM adapter. In this concept, the CSM subsystems support the laboratory module functions for the 120-day resupply period. Consequently, Concept II has a laboratory which is dependent on the CSM subsystems. The third configuration (Concept III) is similar to Concept II with respect to the addition of the separate laboratory module within the LEM adapter section; however, the Concept III laboratory was designed to carry its own subsystems; therefore, it is considered to be independent of the CSM subsystems. The subsystems in the CSM-independent laboratory module were designed for one-year continuous operation with resupply of expendables by the Apollo CSM as required. Included as part of the original Extended Mission Apollo study was the establishment of detailed development plans including costs, schedules and manufacturing and facilities plans. These development plans were based on an integrated but noninterference relationship with the Apollo program and on maximum utilization of Apollo technology, facilities, etc. The laboratory module, for example, utilized a large percentage of Apollo/Saturn interstage tooling.

Results of the initial Extended-Mission Apollo studies revealed several additional factors that warranted further investigation. The required 120-day resupply cycle dictated the use of advanced subsystem concepts in several areas. For example, the existing fuel cell electrical power system in the Apollo was replaced

114PS77263

APOLLO X CONCEPTS

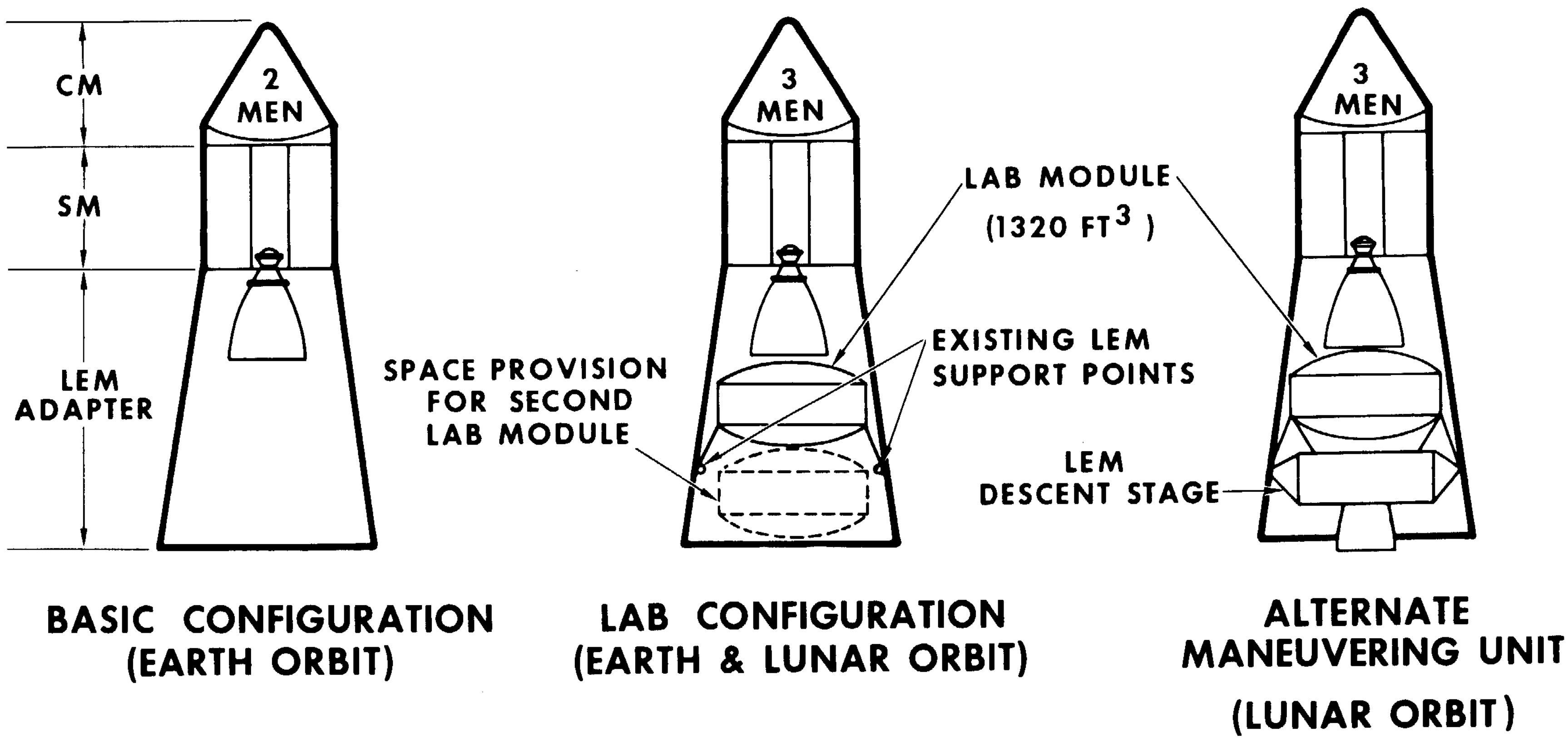


Figure 1

with a solar cell battery power system to eliminate the excessive weight and volume penalties attributable to the fuel cells and associated expendables. Additionally, a molecular sieve was employed in the environmental control system to eliminate the large weight and volume attendant with using the existing Apollo lithium hydroxide CO_2 removal system for 120 days. Consequently, the limiting mission duration which might result from using only current Apollo subsystem concepts was not known. Also, although the feasibility of extending the life of the current Apollo subsystems had been established in the initial study, specific techniques for accomplishing this life extension had yet to be determined.

For these reasons, further studies were initiated as an addendum to the Extended Mission Apollo contract. The purpose of these additional studies was to define the design characteristics and the maximum earth orbital mission duration of the Apollo CSM, assuming restriction to use of existing Apollo subsystem concepts. This configuration was identified as Concept 0. Included in this study was examination of each subsystem to determine the life extensions possible through the addition of spares and redundancies. The study concluded that the earth orbital duration capability of Concept 0 was approximately 90 days (based on Satury IB payload limits).

After review of these findings, NASA initiated the current Extended Apollo Systems Utilization Study, one part of which is to define in depth the design and operational characteristics of a vehicle based on the identical subsystems approach employed in Concept 0, but limited arbitrarily to a 45-day maximum mission duration. This vehicle, identified as Apollo X, appears to represent the most logical next-step in manned spacecraft duration capability beyond the current Apollo and Gemini programs.

As part of this same study program, the characteristics of the CSM configurations associated with the dependent and independent laboratory module (as in Concepts II and III) are being investigated relative to earth orbital and lunar missions of greatly extended durations. Under these Prolonged Mission studies, the Concept II CSM is identified as a Mission Support vehicle, since it supports the laboratory (AORL) by resupplying subsystems, crew members, and expendables. The Concept III CSM is identified as a Logistics Support vehicle since it resupplies only expendables and crewmen, with the CSM subsystems remaining in a quiescent state after docking to the laboratory module. The laboratory modules are being investigated separately (i.e., by another contractor) under a current MSC study program. The definition of the characteristics of the Apollo X laboratory modules is also included as part of this separate AORL study effort.

The primary objective of the Apollo X study has been to define a standard spacecraft design capable of alternately performing extended-duration lunar-earth orbital missions of NASA near-term interest. Included in this primary objective were studies to determine specific modifications required for each subsystem to accomplish the extended missions. Subsystem qualification test programs, which would substantiate the analytically-derived extended-life subsystems, were also defined. In the experimental area, emphasis was placed on definition and integration of experimental packages (for earth-orbital missions) based on experimental lists probided by NASA at the initiation of the study. As an aid to NASA evaluation of the desirability of, or requirement for, the laboratory module, studies were also performed defining development factors based on CSM changes attendant with and without the laboratory module. All development factors investigations were based on an integrated, but noninterference, relationship with the current Apollo program.

The Apollo X study was initially focused toward consideration of earth orbital experimental missions of up to 45 days duration; spacecraft and subsystems requirements were to be developed through the determination of the experimental mission requirements. Secondary studies were aimed at determining the modifications required to the 45-day earth-orbital CSM design when applied to a lunar survey mission (including up to 28 days in lunar polar orbit). Several weeks after study initiation, however, NASA directed that equal emphasis be given to the lunar mapping mission, and requested that primary emphasis be placed on the definition of a standard CSM suitable for both the lunar and earth-orbital missions. Accordingly, a reevaluation was made of previously developed earth orbital mission requirements. This reevaluation indicated that the lunar survey mission imposed overriding requirements in most subsystem areas.

As previously indicated, the overall study approach was based upon the derivation of a standard multimission CSM applied to different vehicle configuration arrangements, as indicated in Figure 2.

In the basic configuration, experimental work space is made available in the command module by elimination of the third crew member and couch and by providing for in-orbit stowage of the center couch under the pilot (left-hand) seat position. This configuration is adaptable to earth-orbital experimental missions, but is not suitable for the lunar missions since it would be required that the service module be completely filled (with propellant, fuel cells, cryogenic tankage, etc.) with no remaining volume available in the SM for mission-oriented equipment. The basic configuration has, however, the definite advantage of being able to provide an experimental laboratory for early flights (such as biomedical and human factors experiments) without schedule dependency on a separate laboratory module. Additionally, these early flights would result in a high confidence base for performing subsequent missions with a laboratory or lunar survey mission module.

The laboratory configuration has increased volume availability as indicated in Figure 2, through the addition of a separate laboratory module carried within the LEM adapter section. The separate laboratory module (approximately 1200-1500 cubic feet volume) is sized by the provision of space inside the LEM adapter for a second module of identical geometry. This configuration can be applied to all of the earth orbital and lunar missions of current interset since mission-peculiar payloads can be installed within the laboratory module. The configuration is extremely versatile in that the experimental volume availability can be controlled by the addition of either one or two laboratory modules. In addition, a single module can be mounted

10-PS7722

SYSTEM EVOLUTION

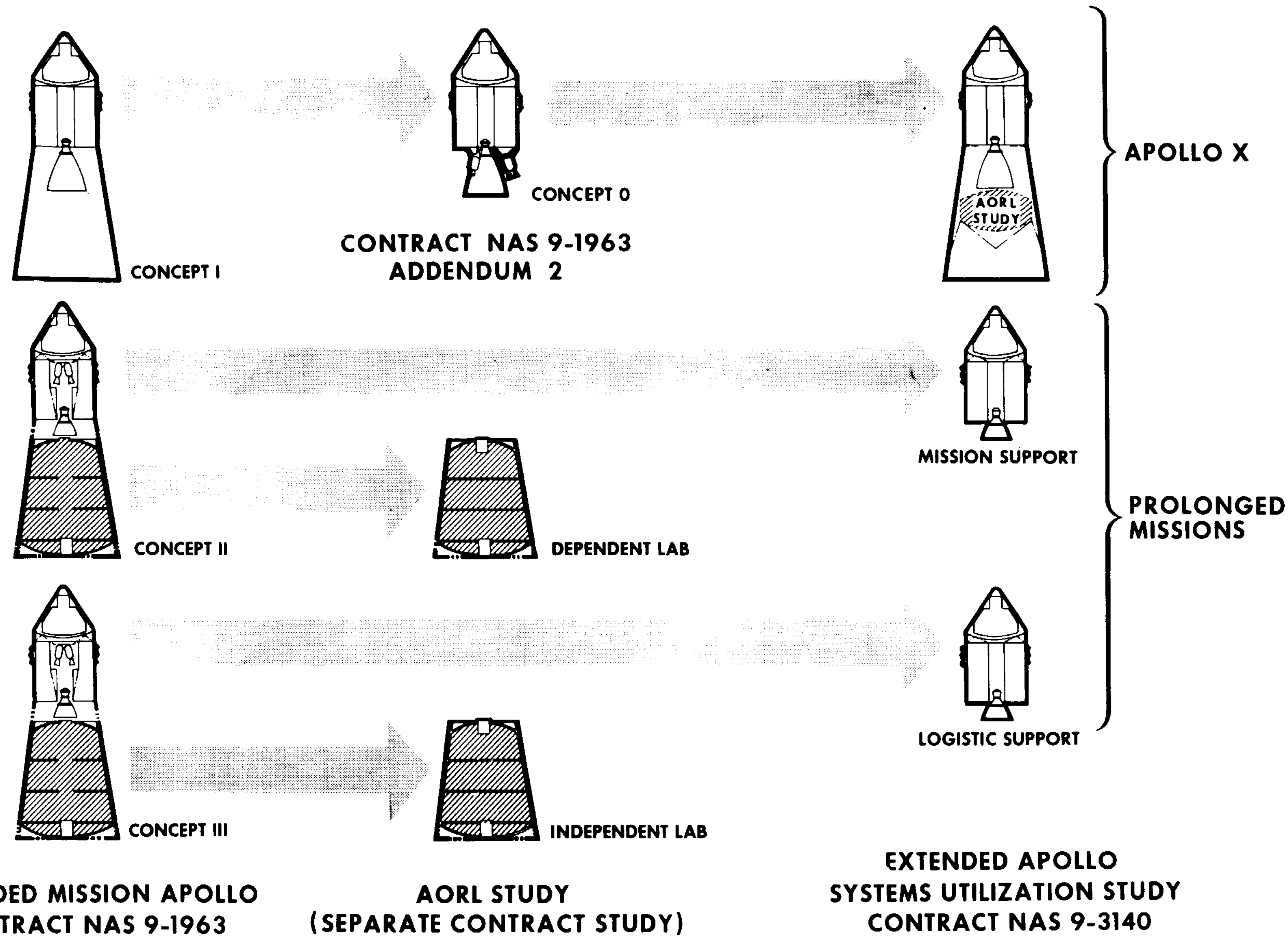


Figure 2

on the LEM descent stage adapter, thereby permitting additional payload for lunar missions, orbital plane change capability, or for providing a lunar landing cargo module.

All technical studies were predicated on the basis of the Apollo Block II configuration. Since past studies revealed that subsystem characteristics materially influenced configuration design, heavy emphasis was placed on subsystems definition as indicated in Figure 3. In this regard, not only primary subsystem components were examined in detail, but secondary components were also fully analyzed. In certain instances where it was found that the addition of spare or redundant Apollo subsystem components might not suffice for the extended-life application, product improvement areas were identified and tentative solutions were established.

Previous studies had identified in-flight maintenance as a major study problem area in view of the number of spares and redundant components used for subsystem life extension. Correspondingly, in-flight maintenance was examined with regard to possible methods of implementing these spares and redundancies as well as determining the overall demands on crew time. These requirements were integrated into crew and experimental station-keeping activities and served as a constraint in the experimental scheduling area. In the experiment design and packaging analyses, it was possible to examine the biomedical and behavioral measurements in great detail through the use of baseline data established in the initial Extended-Mission Apollo study. It is believed that the Apollo X biomedical and human factors experiments as now defined are sufficiently detailed to permit the establishment of equipment specifications.

APOLLO X STUDY APPROACH

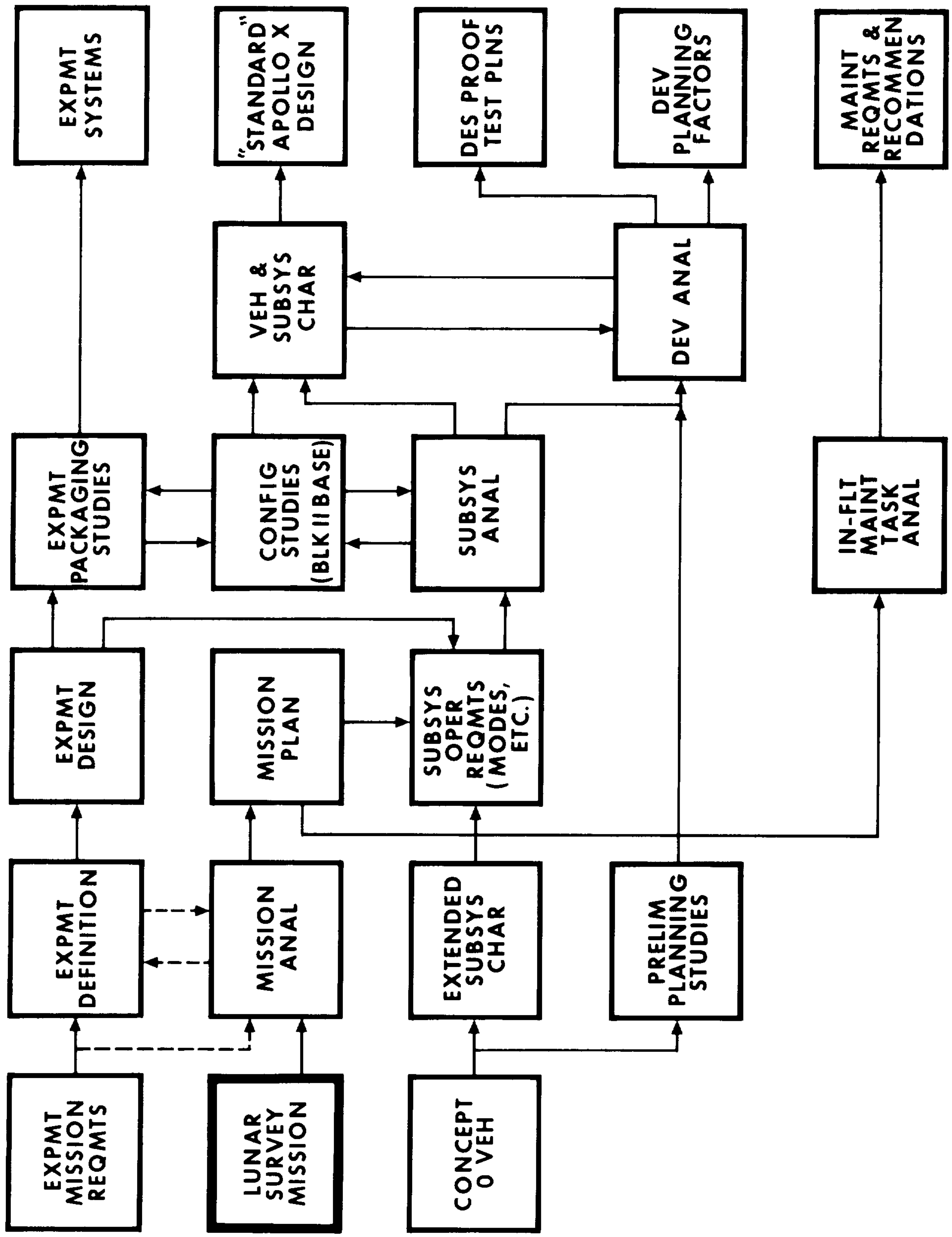


Figure 3

This page intentionally left blank.

CONTENTS

CONTENTS (Continued)

CONTENTS (Continued)

ILLUSTRATIONS

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

TABLES

I. INTRODUCTION AND SUMMARY

12204

This report presents the results of a four month study by The Marquardt Corporation (TMC) of the use of the Lunar Apollo Service Module Reaction Control Engine (SM/RCS) for the Apollo X requirements. This study consisted of prediction of the limiting characteristics of the present engine, identification of performance and environmental areas unique to Apollo X, and design of a developmental plan to assure delivery of a suitable modified engine to meet the program schedule.

Four primary areas of change to meet Apollo X requirements were considered:

1. Change in duty cycle. The duty cycle increase in terms of both burning time and number of starts is well within that already demonstrated with the Lunar Apollo engine. This is not anticipated to be a problem.

2. Extended time in space environment. The more than 100% increase in the time the engine is exposed to a space environment does however introduce several unknowns such as damage from meteorites, combustion chamber coating sublimation, and possible corrosion due to extended immersion of materials in propellants. These problems appear to present the major areas requiring investigation and development.

3. A different thermal radiation profile. The Apollo X thermal environment could be given only cursory evaluation since the Lunar Apollo (Block II) thermal configuration is still in the early stages of development. However, the only question which is left open is whether an active thermal conditioning system is required.

4. Increased reliability allocation. The development and qualification test program in conjunction with a reliability demonstration test phase, are planned to provide assurance that the engine satisfactorily meets its reliability allocation, which is nearly a factor of ten higher than that assigned to Lunar Apollo. Early investigations are planned on components that are critical with respect to reliability in an effort to anticipate developmental requirements in this area.

On the basis of this analysis, the development plans were oriented toward investigation of the technical problems that appear to present potential difficulty in meeting the Apollo X requirements. The results of these laboratory investigations will be incorporated into full scale developmental engines for further evaluation. The resulting modified engine then will be requalified and subjected to a reliability demonstration prior to delivery for spacecraft use. Author

With this approach and considering the well developed state of the Lunar Apollo engine as a point of departure, it appears reasonable that the Apollo X technical objectives and schedules will be met.

II. STATEMENT OF PROBLEM

OPERATIONAL REQUIREMENTS AND DESIGN FACTORS

The operational requirements of the Apollo X missions, as transmitted to TMC, are presented for the Service Module Reaction Control Engine. The primary emphasis has been to define the differences in mission requirements and environments between the Lunar Apollo and Apollo X missions. The relative degree of these differences, in turn, define the basic design factors that were to be analyzed in order to determine the capabilities and limitations of the present engine design for the Apollo X mission.

MISSION PROFILES AND REQUIREMENTS

The following three mission profiles were considered for the Apollo X study:

Profile #1 - Mapping while in lunar polar orbit

Profile #2 - Mapping while in earth polar orbit

Profile #3 - Low inclination earth orbit

Profile #1 was used as the basic design criteria for the Apollo X program. The capability of the qualified Lunar Apollo engine was evaluated for Profile #1 and then its adequacy for the Profile #2 and #3 mission was considered. A detailed description of the above three mission profiles is given in the subsequent paragraphs, along with a comparison of these profiles with the Lunar Apollo mission profile.

Profile #1 - Mapping While in Lunar Polar Orbit

Profile #1 is a 34-day mission; three days in translunar flight, 28 days in lunar polar orbit, and three days in transearth flight. The maximum mission duration for the present Lunar Apollo mission is only 14 days. However, the boost, translunar, and transearth phases were assumed to be similar to that of Lunar Apollo.

Upon reaching the moon, the spacecraft is to be injected into an 80 N.M. lunar polar orbit for the mapping mission. The mapping will be conducted during the 28-day lunar polar orbit phase. The mapping schedule as defined in Table I is subdivided into 110 mapping segments with a spacecraft orientation maneuver required prior to each segment. During the mapping runs, the vehicle is aligned to local vertical with an attitude deadband (all axes) of $\pm 0.5°$ and an allowable rotational rate of $\pm 0.02°$/sec. about two axes and about 3°/min. for the third axis.

TABLE I

MAPPING SCHEDULE FOR 34-DAY
LUNAR POLAR ORBIT MISSION

Operation	Elapsed Time		Cumulative Time	
	Hr.	Min.	Hr.	Min.
Start at Pole	00	00	00	00
Mapping Interval	1	00	1	00
No Mapping	5	15	6	15
Mapping Interval	0	30	6	45
No Mapping	5	15	12	00
Mapping Interval	1	00	13	00
No Mapping	5	15	18	15
	Cycle to 28 days			

NOTE: 330 Lunar Orbits

82.5 hours of mapping

Based upon this information, a mission duty cycle (Table II) was formulated for this mission profile. This mission duty cycle is conservative. The approximately 770 seconds of operation and 10,270 starts calculated for the mission should be a maximum and a significantly lower burning time and number of starts will be required for a nominal mission. The Lunar Apollo mission duty cycle, which was used as the baseline, included both plus and minus pitch engine firings for all rotational maneuvers. The spacecraft orientation maneuver required prior to each mapping segment is assumed to be the same as an average translunar orientation maneuver which entails a heavy spacecraft configuration, and thus requires the greatest burn time and starts for each orientation maneuver.

By way of comparison, Table III is a summary tabulation of the mission duty cycle that is defined in the procurement specification (Reference No. 1) for the Lunar Apollo mission. The only major differences between the Apollo X Lunar Polar Orbit Mapping Mission and the Lunar Apollo mission is the addition of the mapping requirement for the Apollo X.

The Lunar Apollo procurement specification indicates that 5,517 starts and 520 seconds of total accumulated burn time will be required during its specification duty cycle. The Lunar Apollo also has a reliable operating life requirement of 10,000 starts and 1,000 seconds of total accumulated burn time. (NOTE: It should be noted that the reliability goal of 0.997 is for mission duty cycle operation only. The total minimum operating life requirement (with parts replacement, if necessary) for the Lunar Apollo engine is 18,000 starts and 1,800 seconds of total accumulated burn time.)

Figure 4 is a schematic of the vehicle orbit and orientation for lunar polar orbit phase of the mission.

Profile #2 - Mapping While in Earth Polar Orbit

The polar earth orbit mission is designed for a 200 N.M. orbit. The mission duration was assumed not to exceed 45 days duration. The boost and orbit injection profile is shown in Figure 5.

During orbit, mapping operations are to be conducted to the same schedule, requirements, and total time as outlined for Profile #1. The orbit deboost and re-entry phase of the mission is defined in Figure 6.

A detailed mission duty cycle was not formulated for this particular mission as the SM/RCS operation required for most of the maneuvers other than the mapping have not been defined yet. It is safe to assume that the total starts and burn time would be less than for the Profile #1. The basis for this assumption is: (1) the number of starts and burn time for mapping orientation and mapping should be the same as for Profile #1; and (2) the additional SM/RCS maneuvers defined in Figures 5 and 6 do not appear nearly as severe as the combined translunar and transearth phases of Profile #1; and (3) no orbital operations other than mapping orientation or mapping are required.

TABLE II

APOLLO X MISSION DUTY CYCLE
(34 Day Mission Lunar Polar Orbit Mapping Mission)

			With Navigational Sightings and Orientation	
			On Time	Starts
I Translunar Phase				
a) Total for Translunar Phase (same as Lunar Apollo)			230.8	2,298
II Lunar Orbit Phase				
a) Lunar Orbit Acquisition and Orientation	3.074	54		
b) Navigational Sighting Orientation	3.180	95		
c) Navigational Sighting Orientation	17.717	624		
d) IMU Alignment	11.526	215		
* e) Mapping Orientation	331.0	4,630		
** f) Mapping	3.72	310		
g) T.E.I. Orientation	0.485	19		
h) Prop. Settling	22.897	2		
	393.6	5,949	393.6	5,949
III Trans-earth Phase				
a) Total for Trans-earth Phase (same as Lunar Apollo)			144.9	2,023
			769.3	10,270

NOTES: ** 1. Lunar mapping is based on two + pitch engines firing at 0.6 lb-sec. I_t each every 960 sec

2. Lunar Apollo mission is based on + and - pitch engine firings

* 3. Mapping orientation requirements are based on the average Lunar Apollo Navigational Sighting Orientation requirements for the Translunar phase of the mission.

4. Navigational sighting orientations and navigational sighting maneuvers are included for all phases of the mission as specified in the Lunar Apollo procurement specification

TABLE III

LUNAR APOLLO MISSION DUTY CYCLE

(Based on Appendix I of NAA/S&ID Specification MC 901-0004-D, dated August 17, 1964)

TYPE OF MANEUVER	TOTALS			AVERAGES		
	# of Man.	On Time	Starts	On Time Per Man.	On Time Per Start	Starts Per Man.
Translunar Phase						
1) CSM Sep. from S-IVB and LEM	1	19.177	1	19.177	19.177	1
2) Transposition Docking	1	22.272	3	22.272	7.424	3
3) CSM & LEM Separation from S-IVB	1	28.970	1	28.970	28.970	1
4) Navigational Sighting Orientation	30	90.398	1,254	3.013	0.072	42
5) Navigational Sightings	32	21.449	884	0.670	0.024	28
6) ΔV Orientation	3	16.374	81	5.458	0.202	27
7) Propellant Settling	5	28.517	5	5.703	5.703	1
8) Lunar Orbit Orientation	1	3.678	69	3.678	0.053	69
Subtotals	74	230.835	2,298			
Lunar Orbit Phase						
1) Lunar Orbit Acquisition and Orientation	1	3.074	54	3.074	0.057	54
2) CSM-LEM Separation Orientation	1	6.432	73	6.432	0.088	73
3) CSM-LEM Separation	1	19.222	1	19.222	19.222	1
4) LEM Tracking Orientation	1	0578	31	0.578	0.019	31
5) Navigational Sighting Orientation	4	3.180	95	0.795	0.033	24
6) Navigational Sighting	17	17.717	624	1.042	0.028	37
7) IMU Alignment	7	11.526	215	1.647	0.054	31
8) Rendezvous Orientation	1	1.089	52	1.089	0.021	52
9) Maneuver for Docking	1	0.586	29	0.586	0.020	29
10) Lunar Orbit Docking	1	57.320	1	57.320	57.320	1
11) T.E.I. Orientation	1	0.485	19	0.485	0.026	19
12) Propellant Settling	2	22.897	2	11.449	11.449	1
Subtotals	38	144.106	1,196			
Trans-earth Phase						
1) Navigational Sighting Orientation	25	11.358	391	0.454	0.029	16
2) Navigational Sighting	38	13.655	1,148	0.359	0.012	30
3) Proton Event Orientation	2	0.480	23	0.240	0.021	12
4) Proton Event	2	1.115	62	0.558	0.018	31
5) ΔV Orientation	3	5.301	128	1.767	0.041	43
6) Propellant Settling	4	48.892	4	12.223	12.223	1
7) CM-SM Separation Orientation	1	6.455	266	6.455	0.024	266
8) CM-SM Separation	1	57.610	1	57.610	57.610	1
Subtotals	76	144.866	2,023			
Mission Totals	188	519.807	5,517			

VEHICLE ORBIT AND ORIENTATION

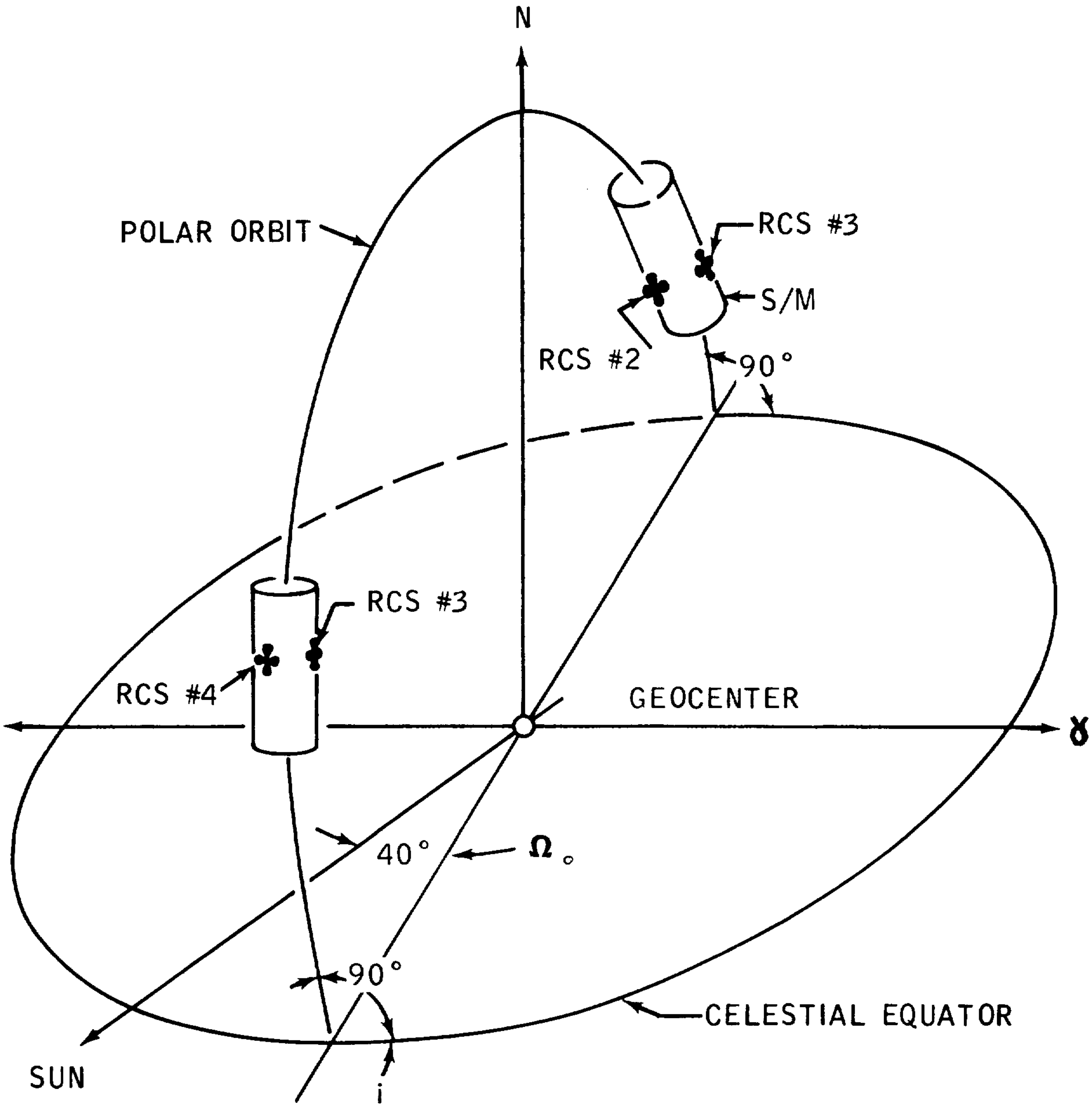


Figure 4

POLAR EARTH ORBIT

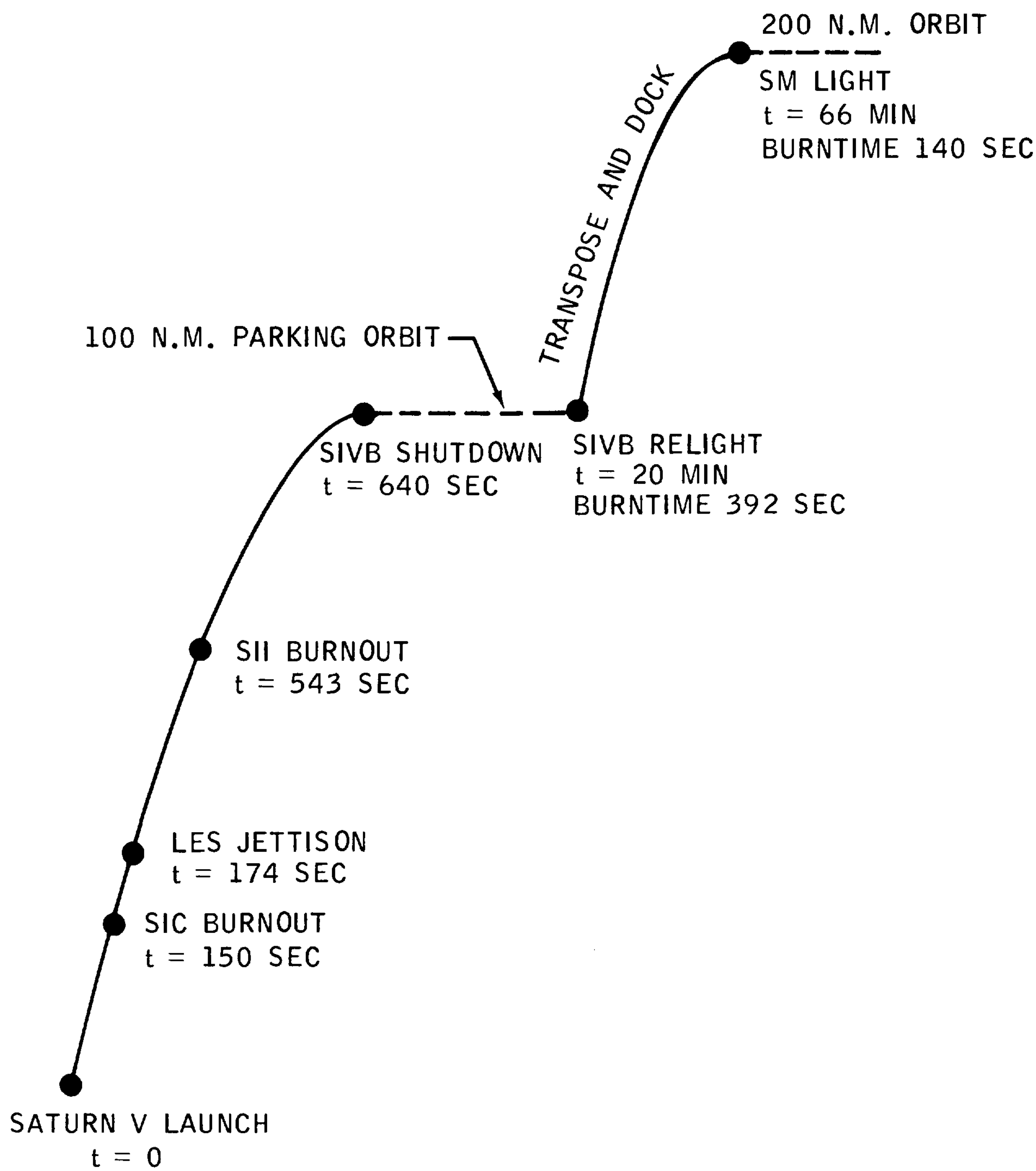


Figure 5

EARTH ORBIT DEBOOST AND REENTRY

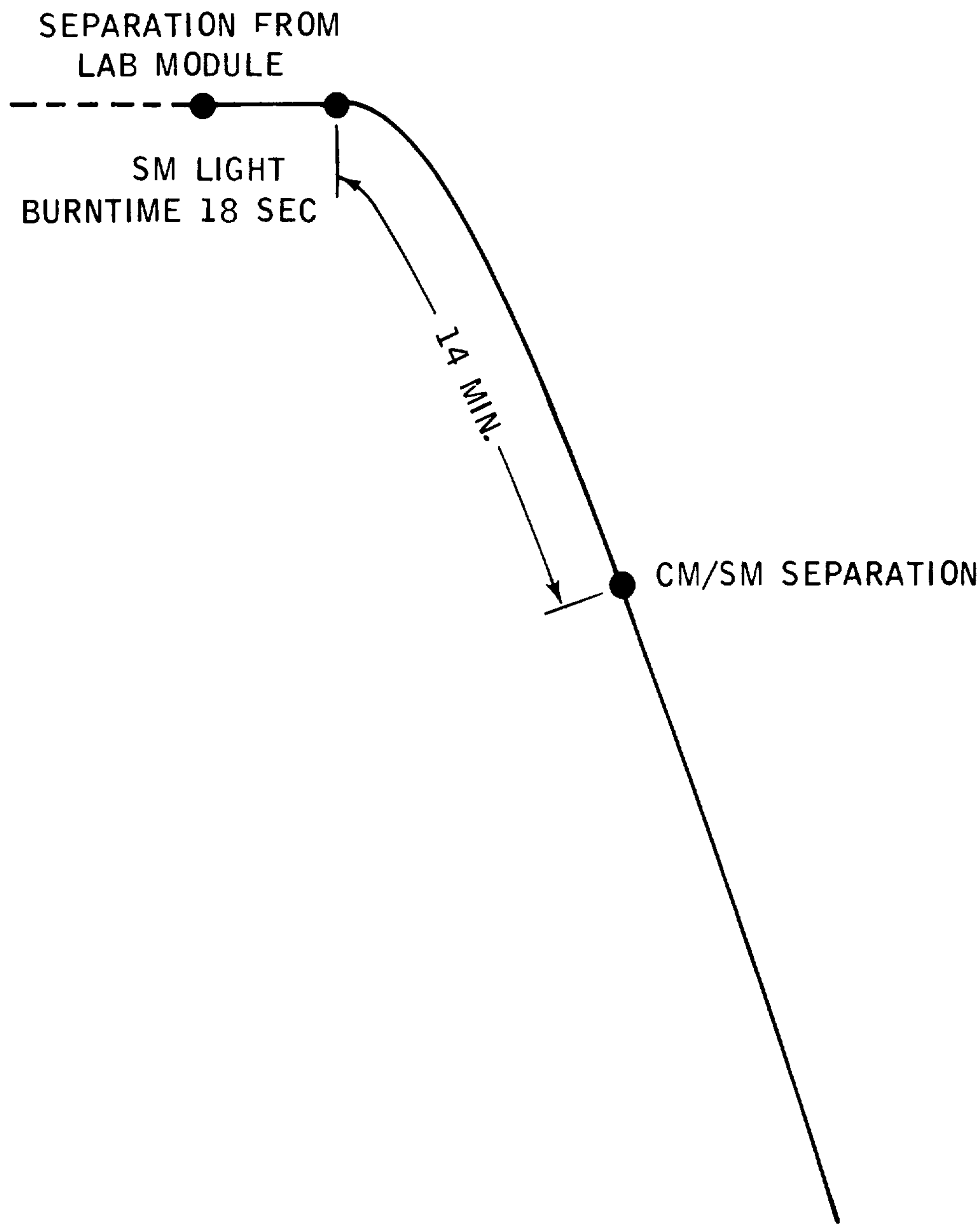


Figure 6

Profile #3 - Low Inclination Earth Orbit

The low inclination earth orbit is designed for a 200 N.M. orbit altitude and 45 days duration. The boost and orbit injection profile is shown in Figure 7. Apparently no SM/RCS orbital operations are required other than to provide an orientation and ullage maneuver at the end of 22 days for an orbit correction by the Service Propulsion System and to provide limited make-up Δ V. The orbit deboost and re-entry phase of the mission is the same as for the Profile #2.

The mission duty cycle has not been formulated for this mission profile because the detailed SM/RCS operation sequence required for the various maneuvers are unknown. However, based upon the above mission profile information, the total starts and burn time for this mission should be significantly lower than even the requirements for the present Lunar Apollo mission.

ENVIRONMENTAL CONDITIONS

It has been assumed that the environmental requirements for: (1) the Ground Handling, Transporation, and Storage Phase; (2) the Prelaunch Phase; and (3) the Boost Loading Phase are the same for the Apollo X missions as for the present Lunar Apollo mission. The detailed environmental requirements for these phases are outlined in the procurement specification (Reference No. 1) for the SM/RCS engines for the Lunar Apollo mission. However, because of the increased mission duration time, space environments such as meteoroids, radiation, etc. which were not included in the specification requirements for the Lunar Apollo mission now become significant factors for the Apollo X mission.

The approach to environmental requirements used in the SM/RCS procurement specification includes two sets of conditions; design environment and test environment. Design environmental requirements reflect the conditions to which the engine might realisitically be subjected in the process of a mission. The specification requires that the engine be capable of satisfactory operation before, during, and after being subjected to these conditions in any natural combination. Test environments are called out as part of the qualification test program and reflect the state of the art for environmental simulation and evaluation.

It was these design and test environments that were examined to determine the adequacy of the present SM/RCS engine in light of the Apollo X mission profiles.

LOW INCLINATION EARTH ORBIT

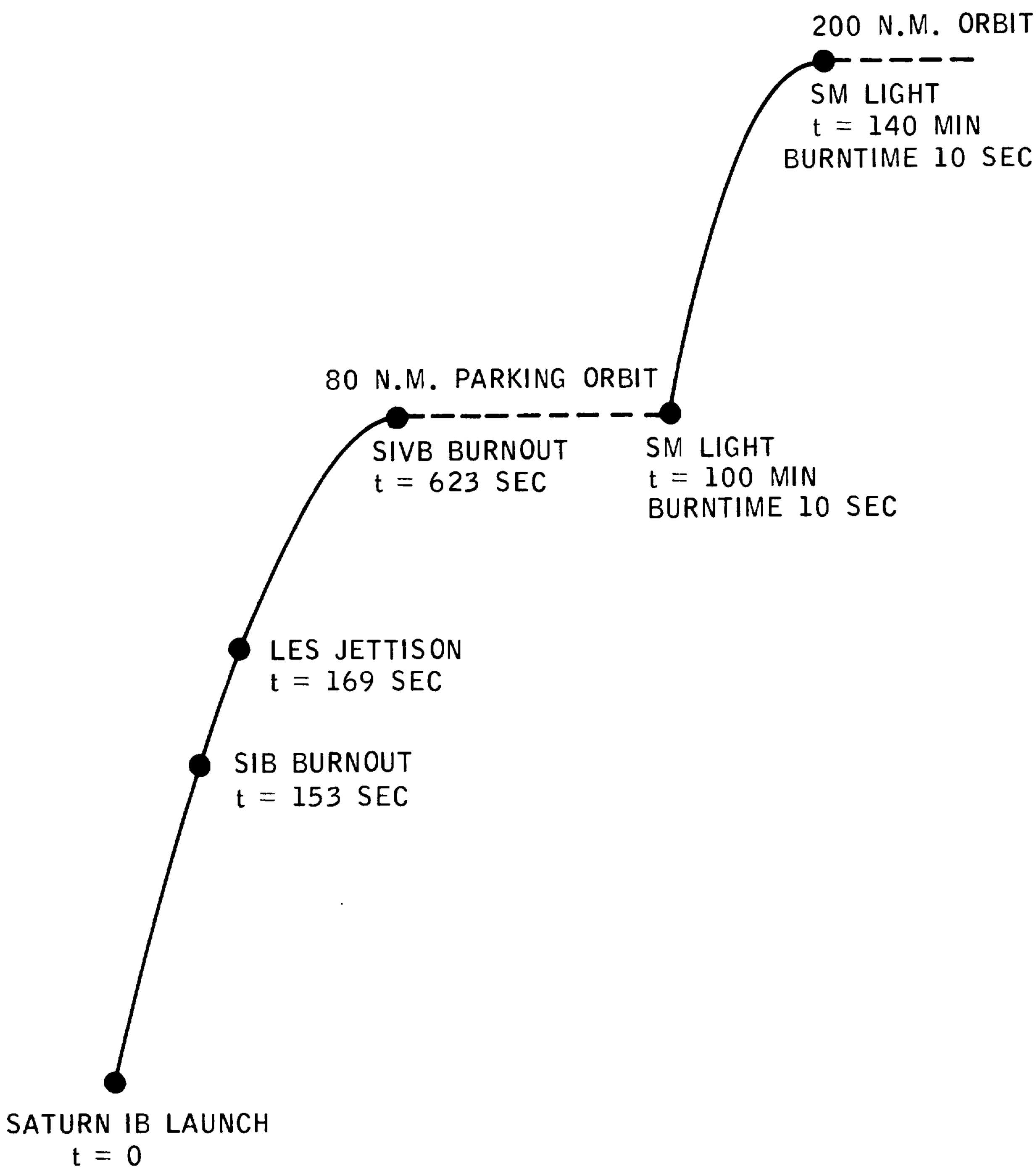


Figure 7

Meteoroids

Meteoroids environment was specifically excluded as a design requirement for the Lunar Apollo SM/RCS engines apparently because of the mission duration. However, with a nearly four fold increase in exposure time, it was decided to assess the severity of this environment as a function of mission time.

NASA/MSC Space Environment Division has recently recommended the use of a specific meteoroid number - mass model from among the many existing models so as to allow comparison of the predictions made by different organizations on the various space vehicle systems and components. Based on these recommendations, the Whipple 1963 number - mass distribution corrected for a meteoroid density of 0.5 grams/cm^3 has been used for both the 200 N.M. earth orbit mission and the 100 N.M. lunar orbit mission. This distribution is presented in Figure 8.

Pressure

The principal concern in the case of pressure environment is the longer periods of time that the engine assembly and components will be subjected to hard vacuum. High vacuum in space can be detrimental to materials in many ways. Among these are: cold welding of metals, evaporation of metals and plastics, surface coating evaporation, and the effect on fatigue and the creep life of metal.

Radiation

The radiation environment was excluded from the required design criteria for the Lunar Apollo mission. Because of the longer mission duration, the effect of this environmental condition becomes more important for the Apollo X mission. Reference No. 2 was used as the principal source of radiation data for this portion of the analysis. Table IV and Figure 9 present typical radiation dosages for a 200 nautical mile earth orbit.

Vibration

Evaluation of the Apollo X mission profile leads to the assumption that the space vibration design requirement will be the same as that for the Lunar Apollo mission. This requirement is that the engine shall sustain a random vibration in three (3) orthogonal axes for a duration of 600 seconds. The random vibration spectrum is as follows:

20 to 100 cps:	Linear increase on a log-log scale from 0.003 to 0.015 g^2/cps.
100 to 200 cps:	Constant at 0.015 g^2/cps.

METEOROID FLUX - MASS DISTRIBUTION

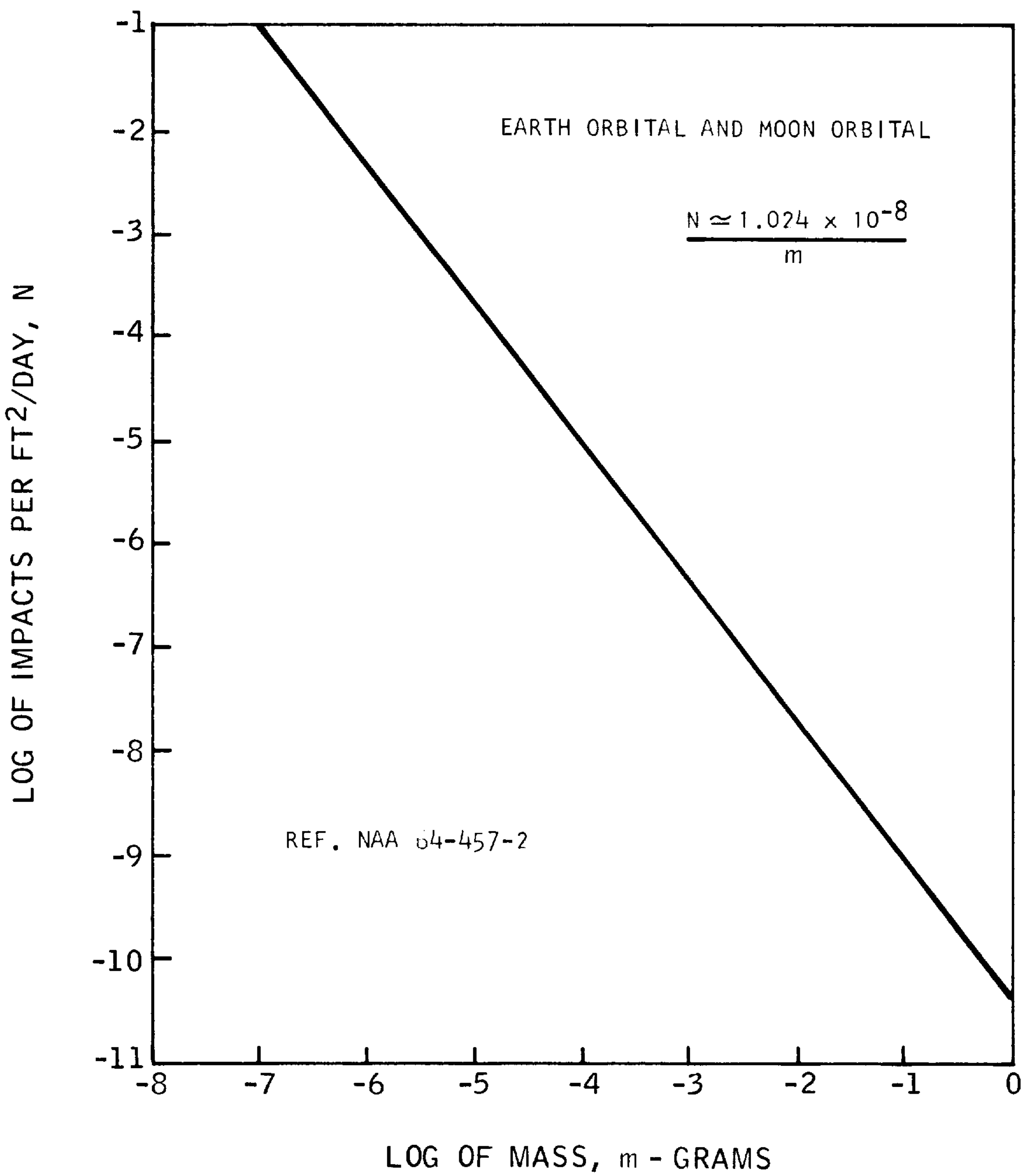


Figure 8

TABLE IV

TOTAL SPACE RADIATION FOR
200 NAUTICAL MILE EARTH ORBIT

Mission Duration	Van Allen Belt + Artificial Elections Aluminum Shielding		Solar Proton Events		
	5 gm/cm^2	1 gm/cm^2	$10^8/cm^2$	$10^9/cm^2$	$10^{10}/cm^2$
45 days	4 - 10	60 - 170	A L L U N D E R 1 . 0		

All energies expressed in Rads (1 Rad = $\frac{100 \text{ ergs}}{\text{gm}}$)

Reference: NAA Report 64-4572, dated 1964

VAN ALLEN BELT RADIATION
PROTONS + ELECTRONS
DELIVERED TO ALUMINUM SHIELDING (1g/cm^2)

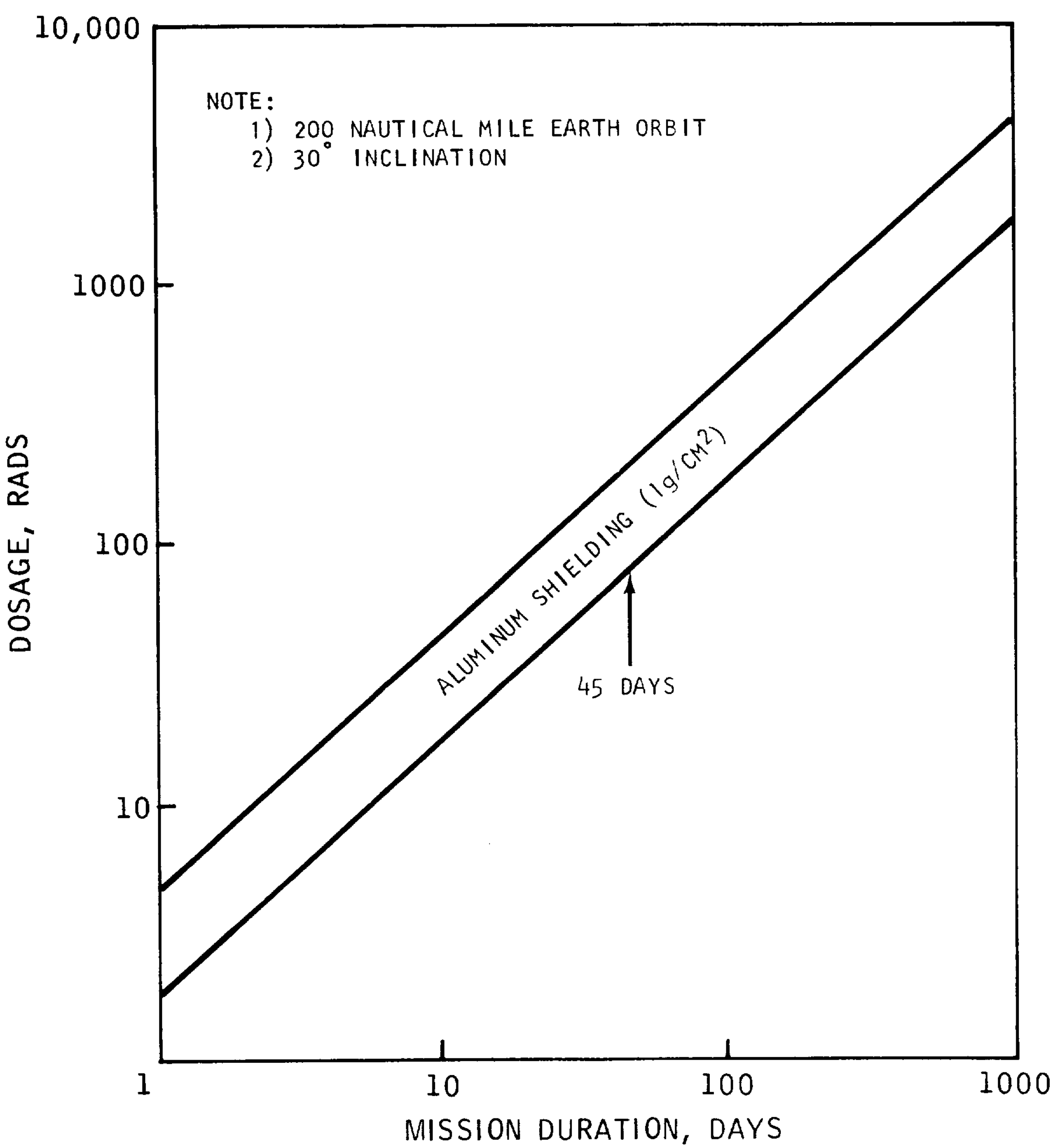


Figure 9

Thermal Environments

Even though the basic environments are essentially the same for both Lunar Apollo and Apollo X, changes in the mission profile, mission duration, orbit inclination, and vehicle attitude necessitated that a thermal study be conducted to assess the effects of these changes.

The lunar polar orbit with a 75° ascension angle was considered by NAA/S&ID and TMC to be the worst case from a thermal standpoint. The attitude orientation for the spacecraft during the above orbit is shown in Figure 10. Figures 11 through 14 define the incident thermal radiation characteristics for each of the four (4) SM/RCS panels. The absorbed radiation is for the SM/RCS panels and is not equal to that for the SM/RCS engine assembly because of the differences in thermal radiation properties of the materials. The predicted panel surface temperatures for a 72° ascension angle for both the earth and lunar polar orbits are shown in Figures 15 and 16 respectively.

Propellant Exposure

Propellant exposure and material compatibility considerations becomes a significant design factor because of the extended mission exposure time. Since propellant compatibility problems and corrosion are dependent upon factors such as temperature and space radiation, as well as exposure time, detailed analysis of these variables were required.

Combined Environments

Table V points out the various combined environmental variables to which the Apollo X hardware will be exposed. The simultaneous exposure to combined environmental effects for long time durations is in general quite detrimental to many nonmetallic materials. Unfortunately, the possible interactions and total degradation or properties cannot be calculated or determined, in general, from existing data concerning individual environmental conditions. Relatively little combined environment evaluation testing has been conducted during the Lunar Apollo SM/RCS engine program. The longer mission times plus the necessity for inclusion of previously ignored environments (i.e., radiation, meteoroids) into the design criteria necessitates that a close assessment be made of the potential effects of combined environments.

Reliability

A tabulated comparison of the reliability requirements for the various mission profiles is as follows:

LUNAR APOLLO:	0.997 (Goal)
APOLLO X	
Lunar Polar Orbit:	0.99919
Earth Polar Orbit:	0.99973
Earth Low Inclination:	0.99997

RCS ORIENTATIONS

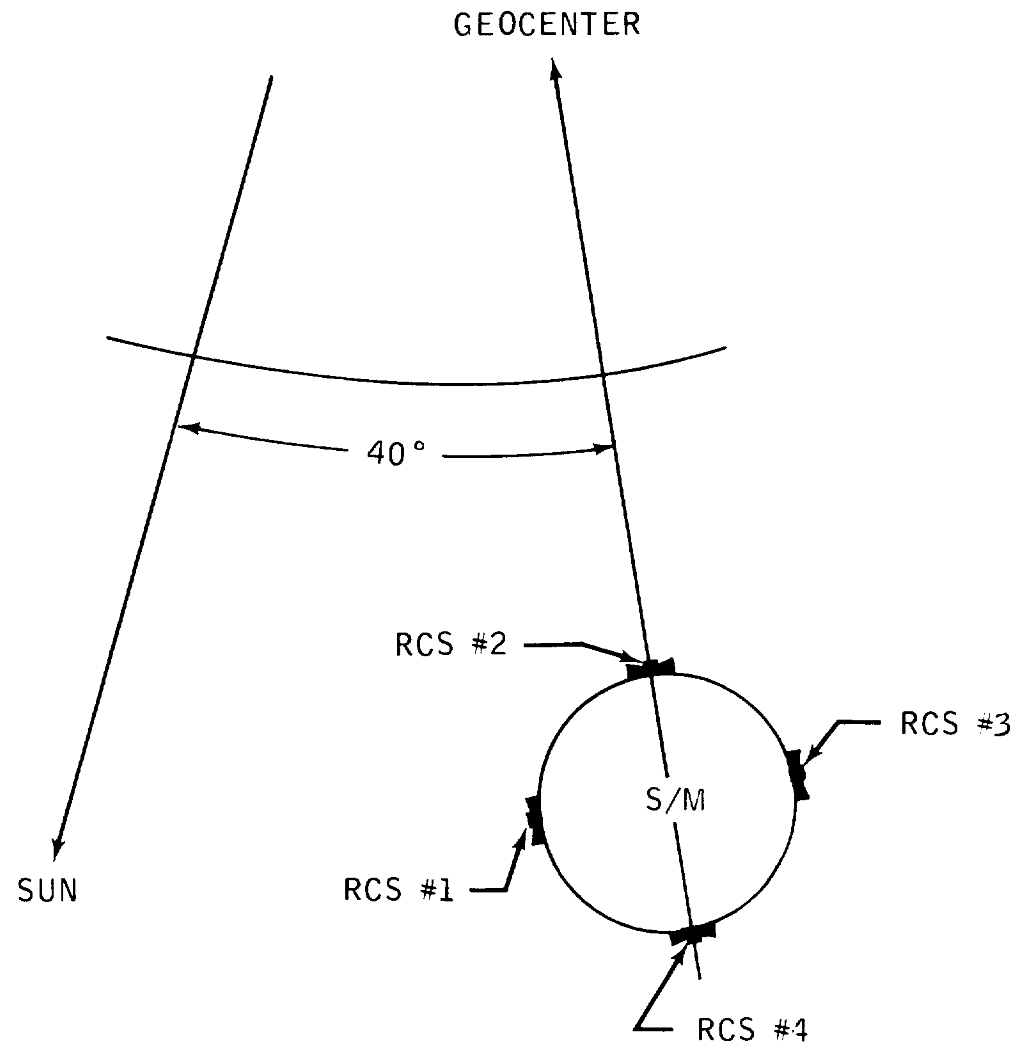


Figure 10

INCIDENT RADIATION VS ORBIT POSITION

RCS PANEL #1

LUNAR POLAR ORBIT - 75° ASCENSION ANGLE

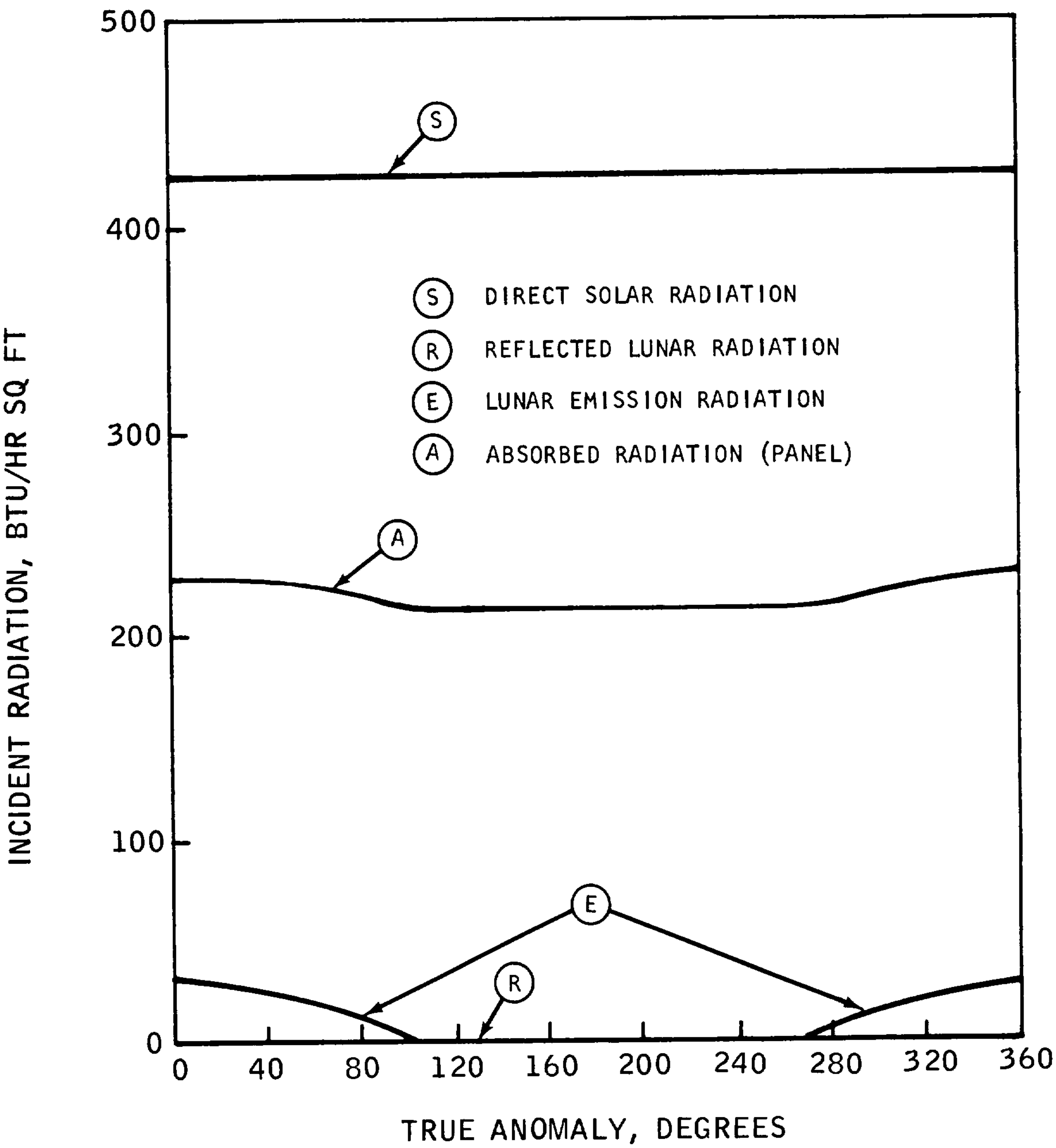
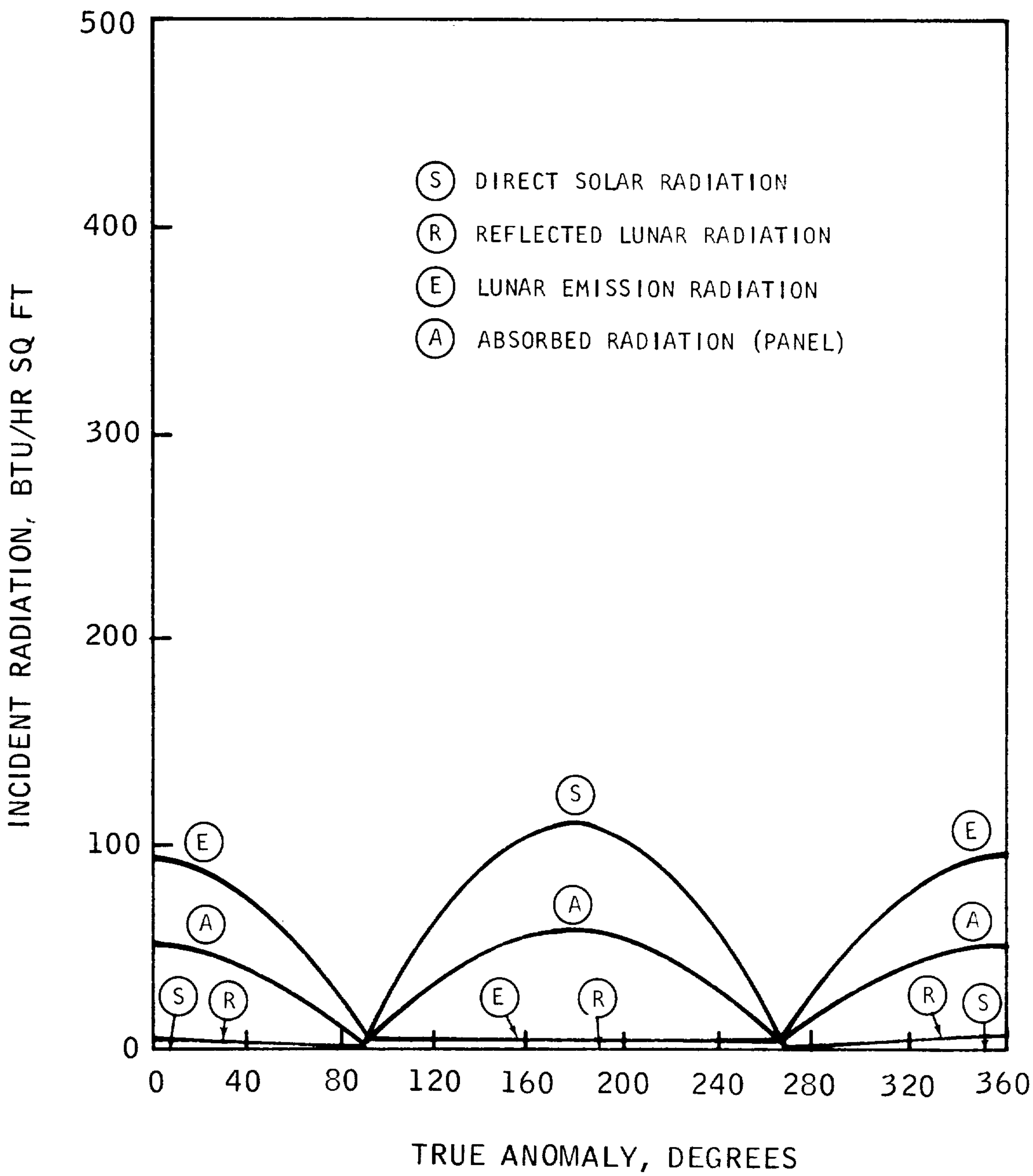


Figure 11

INCIDENT RADIATION VS ORBIT POSITION

RSC PANEL #2

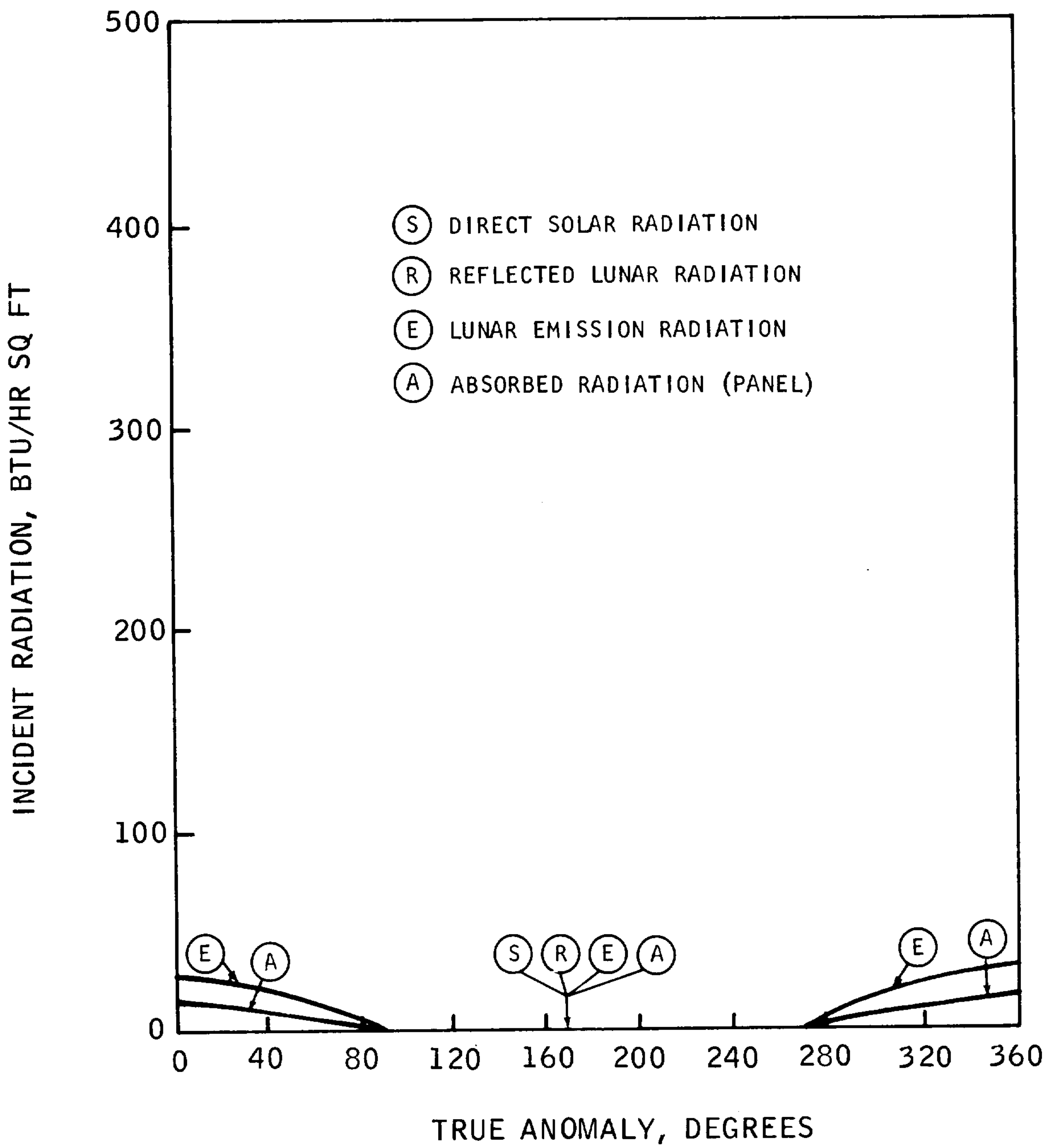
LUNAR POLAR ORBIT - 75° ASCENSION ANGLE



INCIDENT RADIATION VS ORBIT POSITION

RSC PANEL #3

LUNAR POLAR ORBIT - 75° ASCENSION ANGLE



INCIDENT RADIATION VS ORBIT POSITION

RSC PANEL #4

LUNAR POLAR ORBIT - 75° ASCENSION ANGLE

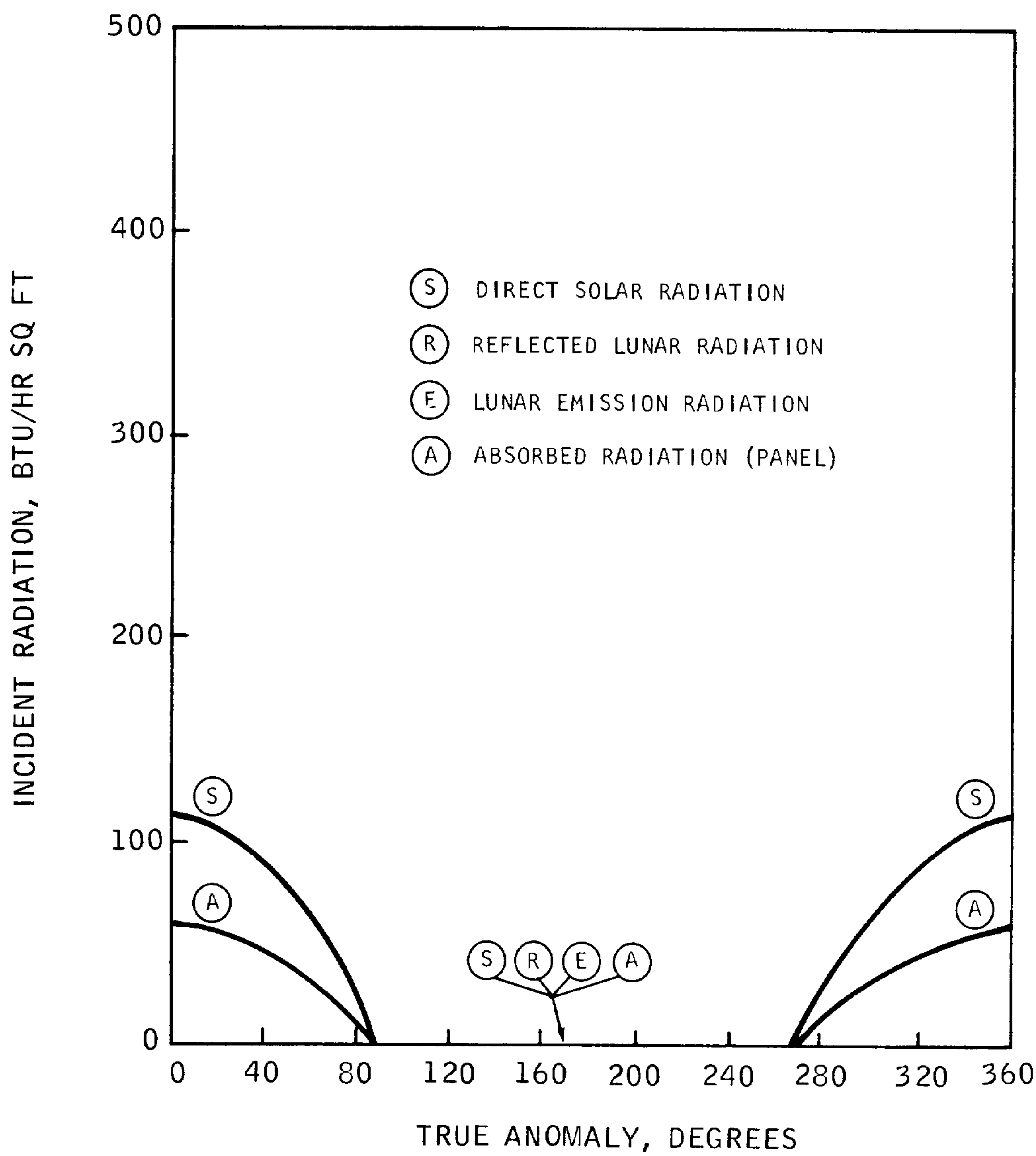
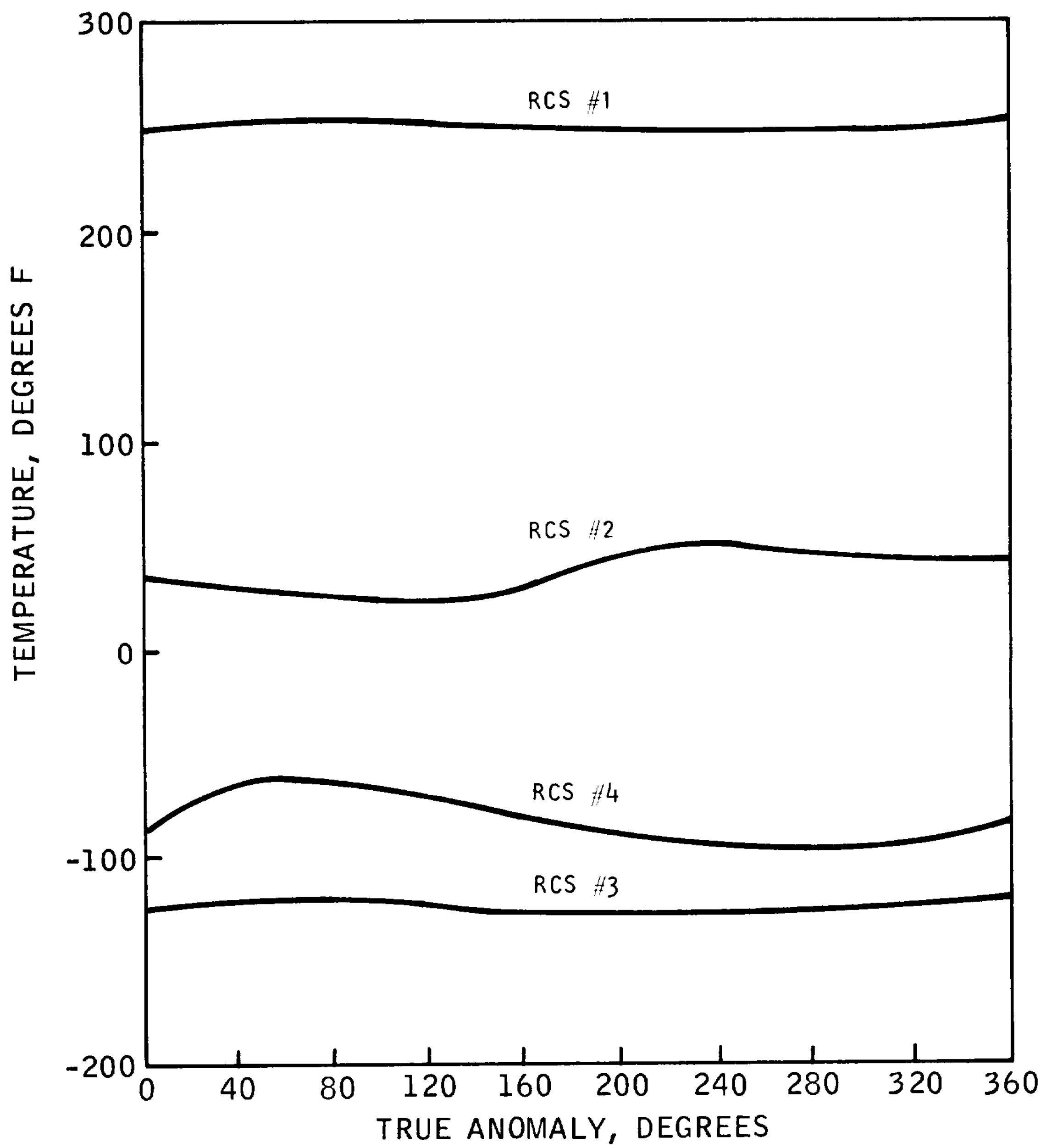


Figure 14

SM/RCS PANEL TEMPERATURE VS ORBIT POSITION

EARTH POLAR ORBIT - 75° ASCENSION ANGLE



SM/RCS PANEL TEMPERATURE VS ORBIT POSITION
LUNAR POLAR ORBIT - 75° ASCENSION ANGLE

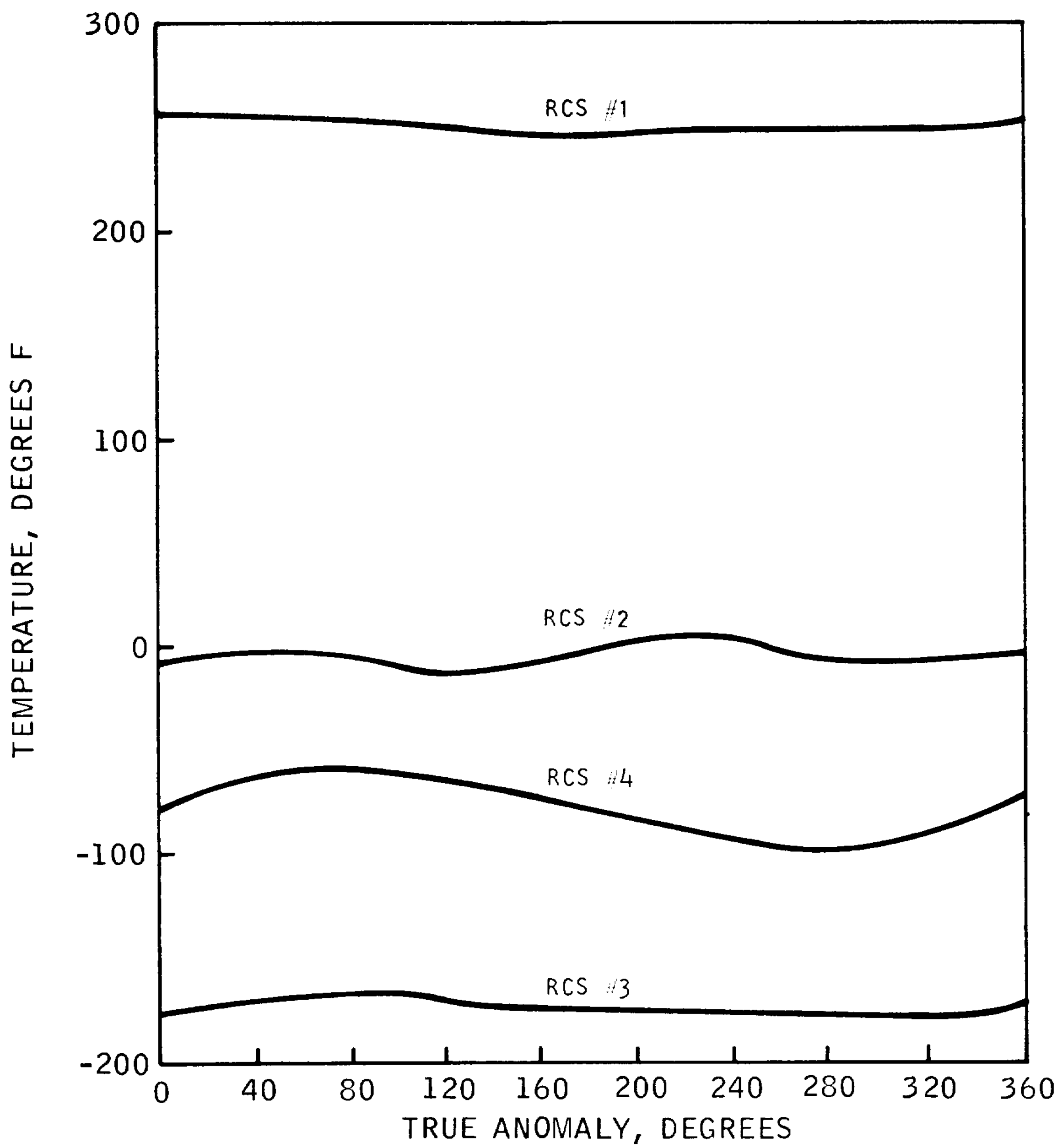


TABLE V

COMBINED SPACE, ENVIRONMENT AND TIME EFFECTS ON APOLLO MATERIALS

Item	Temperature	High Vacuum (evaporation)	"Nuclear" Radiation	Compatibility and Corrosion	Cold Flow of Stressed Plastics	Cold Welding of Metals
Propellants	×		×	×		
Magnetic Metallic Components (Solenoid Valves)			×			
Metals			×	×		×
Phenolics - Glass Reinforced		×	×		×	
Durak B Coating on Unalloyed Molybdenum	×	×	×			
Seals						
a. Teflon (TFE) (P_c Tap, Omniseal cover)	×	×	×	×	×	
b. Teflon - glass filled (valve seat)	×		×	×	×	
c. Teflon (FEP) coating (valve body seal)	×	×	×	×	×	
Electrical Insulation						
a. PYRE-ML Coating (coil wire)	×		×			
b. 3M #250 epoxy resin (coil impregnation)	×	×	×			
c. 3M #248 epoxy resin (coil potting)	×	×	×			
d. Tape - polyethylene	×	×	×			
e. Tape - polyester 3M #74	×	×	×			
f. Tape - glass	×	×	×			
g. Teflon (FEP) (grommet)	×	×	×			
h. Teflon (cable insulation)	×	×	×			

× - indicates exposure

It should be noted that the reliability requirement for the Lunar Apollo mission is a goal and no reliability documentation test phase is presently included in the Lunar Apollo Program to demonstrate this reliability.

As a result of the recent redirection of the SM/RCS engine program the Qualification engine will not be finalized until later this year. Therefore, no data is presently available to calculate a valid reliability estimate on the final Qualification engine configuration. Limited extrapolations can be accomplished based on previous component and engine data of a different configuration.

DESIGN CONSTRAINTS AND ASSUMPTIONS

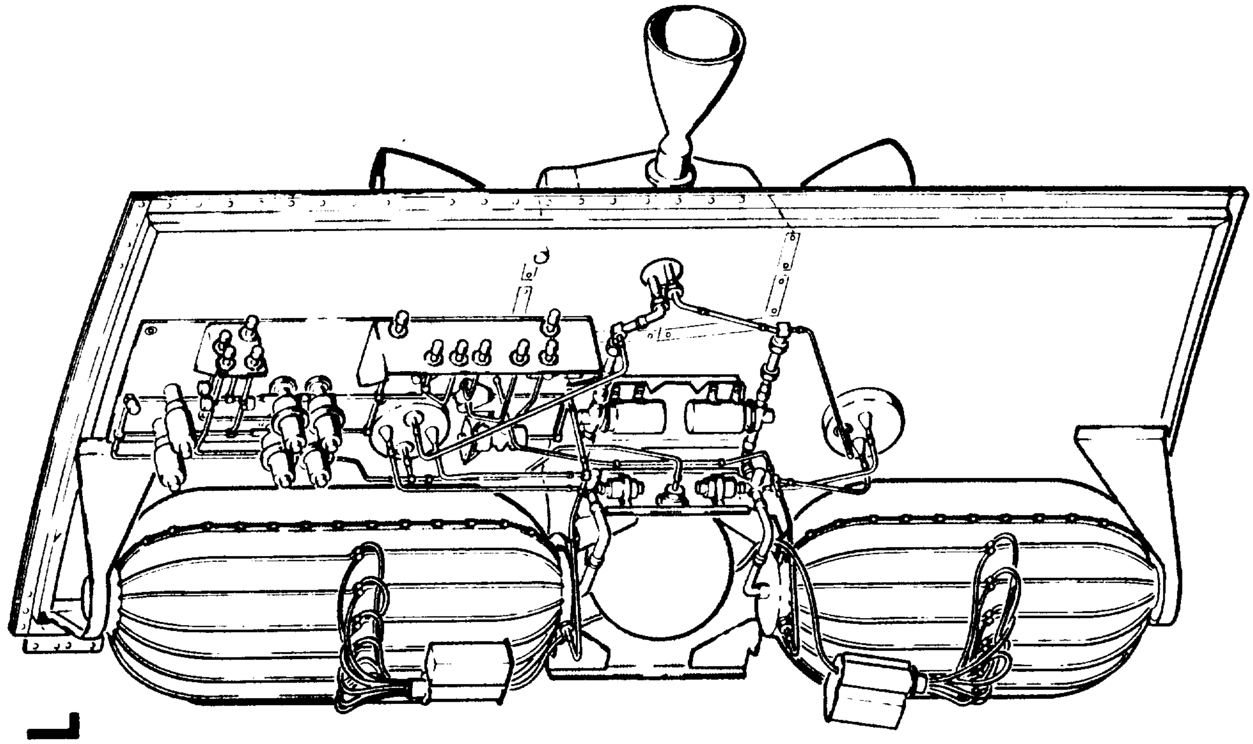
This section outlines the basic assumptions and design constraints that were utilized as the baseline and limiting conditions from which the detailed study program was conducted to determine the adequacy of the Lunar Apollo SM/RCS engine for the Apollo X mission.

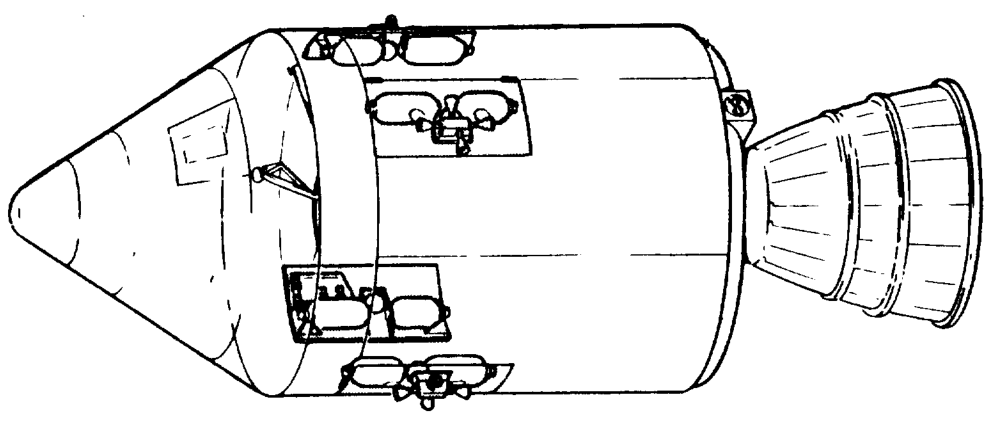
The section on interface definition defines the basic inputs and system interfaces that have been established during the Lunar Apollo Program. Some of these areas are still in a state of "flux" because the Block II design has not been completely finalized. Therefore, the interfaces that were valid at the start of this study were used.

The design constraints outlined should not be considered as necessarily mandatory, but only as an arbitrary baseline and limiting boundary upon which this study was conducted. The development program has demonstrated that the engine assembly is probably capable of operating over a wider range of conditions and environments. Thus, if system or mission trade-off analyses indicate that changes in engine conditions are desirable, additional documentation testing is likely to reveal that little or no compromise will result in engine reliability or operating characteristics.

PROPELLANT SYSTEM

It is assumed that the propellant supply system for Apollo X will be the same as that for Lunar Apollo or will contain only minor changes. Figure 17 shows the SM/RCS panel assembly that contains the SM/RCS engines. A schematic of the propellant system is shown in Figure 18. The propellant is supplied to the engine inlet at a nominal static pressure of 181(+9,-4) psia, and a dynamic inlet pressure (flowing) of 170(+9, -4) psia. A relief valve is installed in the system to insure that the static inlet pressure will never be greater than 250 psia. The temperature of the propellant at the engine inlet will be within the range of 40 to 100°F.





S/M REACTION CONTROL SYSTEM

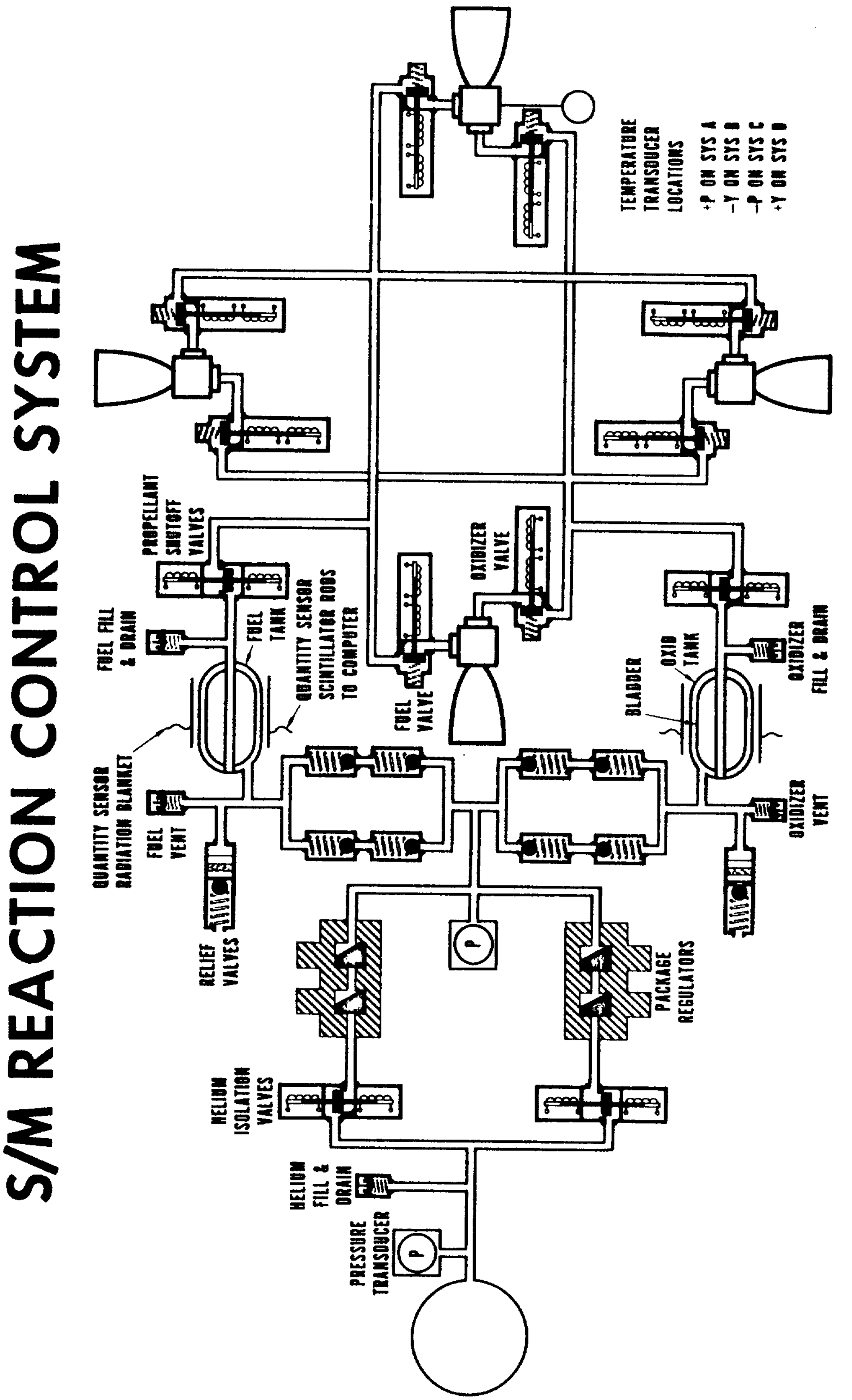

S/M REACTION CONTROL SYSTEM
TEMPERATURE TRANSDUCER LOCATIONS
+P ON SYS A
-Y ON SYS B
-P ON SYS C
+Y ON SYS D
PROPELLANT SHUTOFF VALVES
OXIDIZER VALVE
FUEL FILL & DRAIN
FUEL TANK
QUANTITY SENSOR SCINTILLATOR RODS TO COMPUTER
FUEL VALVE
OXID TANK
BLADDER
OXIDIZER FILL & DRAIN
QUANTITY SENSOR RADIATION BLANKET
FUEL VENT
OXIDIZER VENT
RELIEF VALVES
PACKAGE REGULATORS
HELIUM ISOLATION VALVES
HELIUM FILL & DRAIN
PRESSURE TRANSDUCER

Figure 18

ELECTRICAL CONTROL SYSTEM

It is assumed that the interface inputs between the SM/RCS rocket engine, the stabilization and control system, and the guidance and navigation system will be the same for both the Lunar Apollo and the Apollo X missions.

The electrical input to the automatic coil of the engine valves is supplied by the reaction jet solenoid driver circuit shown in Figure 19. The automatic coils of the fuel and oxidizer valves of a single engine are connected in parallel to the solenoid driver for each engine.

The direct coils of the fuel and oxidizer valve of a single engine are connected in series. The electrical input to the direct coil is supplied through microswitches in either the rotational or translational hand controllers.

The safe operating voltage range for the solenoid valves is 21 to 32 volts d.c., and the engine must perform properly between 24 and 30 volts d.c. The automatic coil operation will not require more than 2.0 amperes per engine at 27 volts d.c. and the direct coil operation will not require more than 1.0 ampere at 27 volts d.c.

THERMAL CONTROL SYSTEM

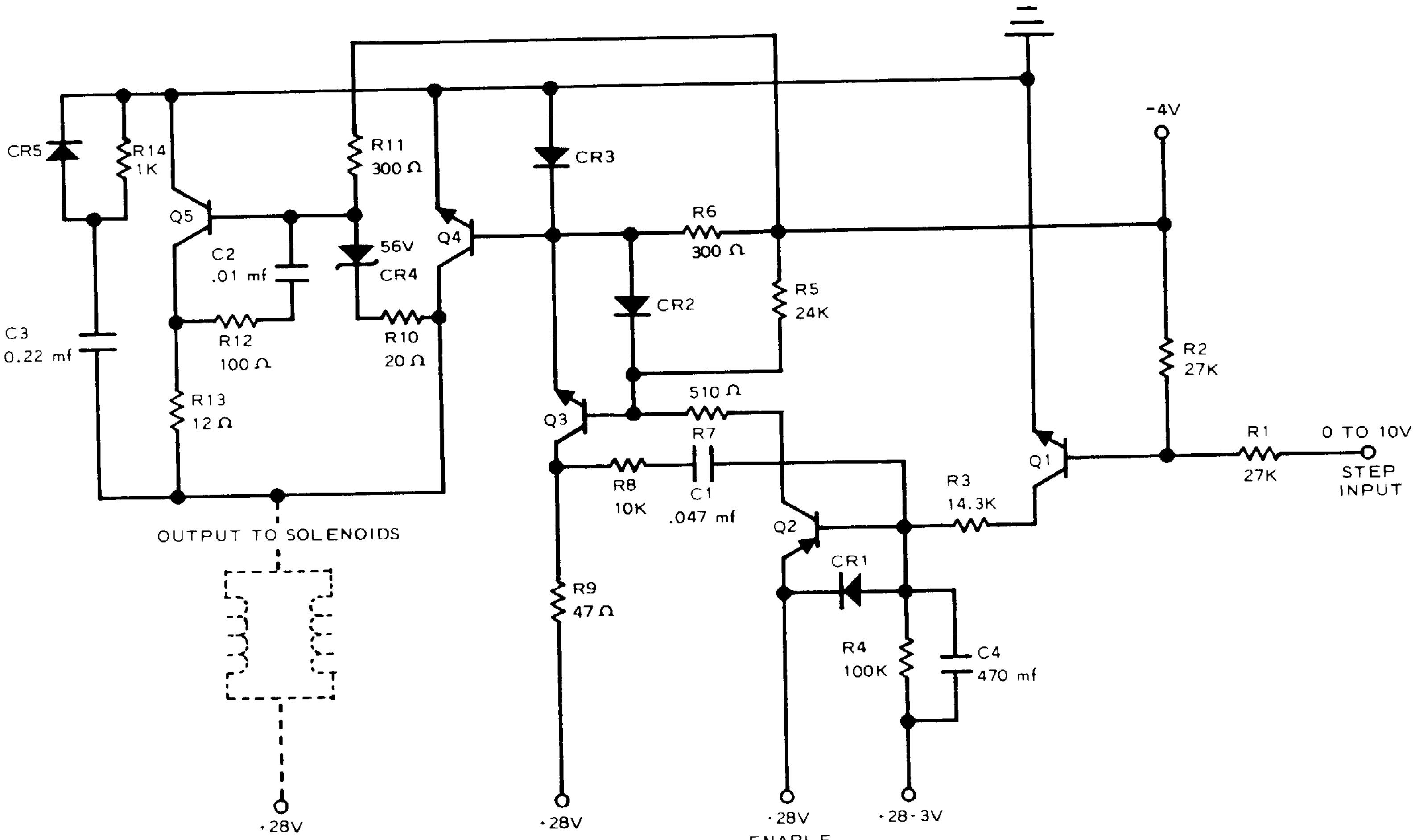
At the start of the study contract, it was specified by NAA/S&ID that the assumption should be made that active thermal control (water-glycol and heaters) would be utilized in Apollo X as in Apollo Block I. However, since that time the thermal control system concept for the SM/RCS for both Block I and II has been under detailed evaluation and study by NASA and NAA/S&ID. Thus, the final definition of the system was in a state of "flux" throughout this study program. Therefore, it was assumed for the purposes of this study that the system contractor would provide a thermal control system for the Apollo X mission which would maintain the inlet propellants, valve, and injector head temperatures within the present requirements of the Lunar Apollo procurement specification (Reference No. 1).

ASSUMPTIONS

a. General

The point of departure for this study was the qualified Lunar Apollo SM/RCS engine. However, the one anomaly in this approach is that the engine configuration, (P/N 227486) which are available as a model for this study, will not be subjected to a Qualification Test. This is because of the recent redirection by NAA/S&ID requiring extension of the development cycle in order to incorporate passive thermal control features in the engine. It is this forthcoming configuration that will be subjected to qualification testing.

DRIVER CIRCUIT FOR SM/RCS SOLENOID VALVES



TRANSISTORS

Q1	2N720A – FAIRCHILD	
Q2	2N1132B+ – HUGHES – SEMI-CONDUCTOR	
Q3	2N2633 – HONEYWELL	
Q4	2N1724 –	TEXAS INSTRUMENT
Q5	2N1724 –	

DIODES

CR1	IN647
CR2	IN647 – PACIFIC SEMI-CONDUCTOR INC
CR3	IN647
CR4	IN29998 – HOFFMAN
CR5	IN3189 – MOTOROLA

Figure 19

It has also been assumed that the final qualified engine will be completely satisfactory for the Lunar Apollo mission and that no problems will be experienced because of the present lack of knowledge or unknown effects of either system or space flight operation.

ENGINE DEFINITION - (LUNAR APOLLO BLOCK II)

At the time this study program was initiated, an engine incorporating a preigniter was undergoing final development and it appeared that the Prequalification Test of that engine would be completed prior to the completion of the Apollo X study. The Qualification Tests were to follow directly. Thus, the Apollo X study program was directed toward using that final preigniter engine design (P/N 227486) as the baseline for the evaluation. In late September TMC was directed to develop a semi-passive thermal control engine. Because of this redirection, a Preliminary Flight Rating Test (PFRT) Program is being conducted on the P/N 227486 engine in lieu of the previously scheduled Prequalification Test Program. The purpose of the PFRT is to qualify this engine for interim usage on Block I Apollo flights. In the meantime, a comprehensive design and development program has been initiated to produce the passive thermal control engine. A major configuration differences in the passive thermal control engine are the incorporation of a thermal standoff of the fuel valve and possibly a higher thermal resistance seal between the Injector Head and the chamber. A Qualification Program will be conducted on the finalized passive thermal control engine beginning in February 1965. Therefore, the Apollo X study program has by necessity been based on the PFRT engine. It is not anticipated that the engine characteristics will change markedly with the introduction of these configuration changes except of course in its thermal behavior.

ENGINE DESCRIPTION

The SM/RCS engine produces 100-pounds vacuum thrust. It is a pressure fed, liquid bipropellant, radiation cooled engine. The hypergolic propellants used are nitrogen tetroxide (N_2O_4) and a 50-50 mixture of hydrazine (N_2H_4) and unsymmetrical dimethyl hydrazine (UDMH). They are injected at an oxidizer to fuel ratio of 2.0 by weight during steady state operation. The engine assembly is capable of either steady state (continuous) or pulse operation.

The basic components of this engine are the fuel solenoid valve, the oxidizer solenoid valve, the injector head assembly, and the combustion chamber and exit nozzle. Figure 20 shows the physical relationship of these components.

LUNAR APOLLO SM/RCS ENGINE

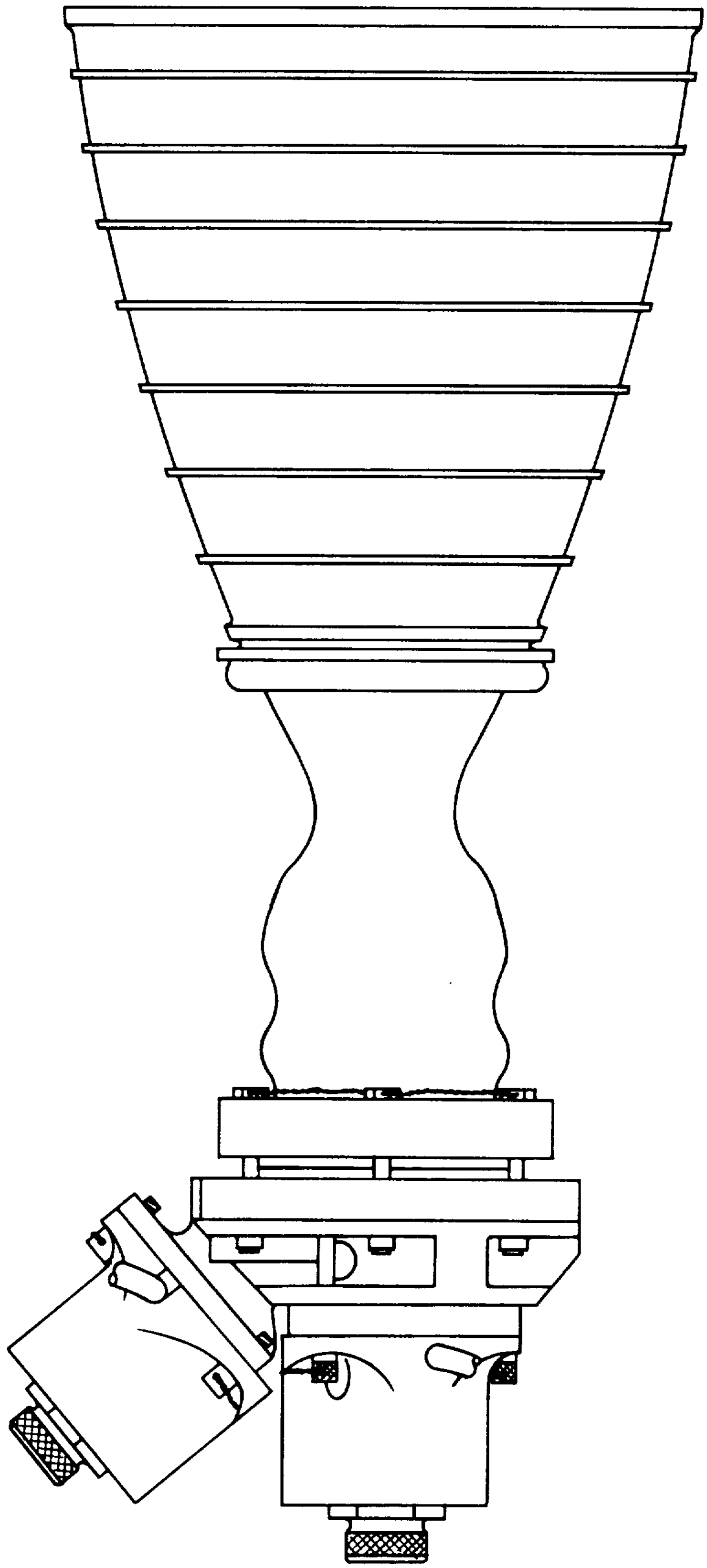


Figure 20

Solenoid Valves

The function of the solenoid valve is two-fold. It must respond rapidly to an electrical open or close command. Secondly, it must provide a positive propellant shutoff to prevent leakage during periods of engine non-operation.

These valves are of the poppet type with a conical seat of teflon and metal. The actuator is the dual coil coaxial solenoid type. One coil is used for automatic control operation, while the second coil is used in the event a manual override is required. They are normally closed valves, designed to "fail-safe" in the closed position. Integral fixed orifices are incorporated into each valve to provide the correct flow rate and oxidizer-fuel ratio at the design inlet pressures.

The automatic coils of the fuel and oxidizer valves are in parallel electrically and designed so that the fuel valve will open prior to the oxidizer valve at nominal operating conditions, thus producing a fuel lead into the combustion chamber. The direct coils are connected in series, and the fuel valve opens approximately 9 ms earlier than the oxidizer valve at nominal operation conditions.

Injector Head Assembly

The injector head introduces and mixes the propellants in the combustion chamber in such a manner that ignition occurs and stable combustion results with a minimum chamber wall temperature.

Initial combustion occurs in the preigniter chamber. The hydarulic passages of the injector head are designed such that the propellants arrive at the preigniter prior to arriving at the main chamber doublets. Thus, ignition will occur in the preigniter chamber with a resultant pressurization of the main chamber prior to the arrival of propellant from the main doublets. It is pre-pressurization of the chamber that produces smooth ignition characteristics of the main doublets without any significant overpressure condition.

The main chamber injector consists of doublets of the unlike impingement pattern.

The combustion chamber is film cooled by injecting propellant along the chamber walls. The propellant subsequently evaporates from the walls and enters into the main combustion reaction.

Combustion Chamber

The combustion chamber consists of two parts; the combustor, and the expansion bell. The combustor is unalloyed molybdenum of ribbed construction which has a molybdenum disilicide coating. At an expansion ratio of 6.9, an expansion bell is attached. The expansion bell is fabricated from L-605 (a nickel alloy) and is joined to the combustor by a Waspalloy nut. The chamber operates at a nominal pressure of 96 psia. The expansion ratio of the combustion chamber is 40 to 1.

PERFORMANCE

Nominal I_{sp} for the PFRT engine at the 5.0 second data point from engine start is approximately 286 seconds.

Figure 21 presents the nominal pulse performance characteristics (I_{sp}, I_t, and O/F) as a function of electrical pulse width. This data is for nominal conditions and an off time of greater than 400 milliseconds between pulses.

OPERATING TEMPERATURES

The nominal steady state operating temperatures of the combustion chamber are approximately 2250°F at the throat and less than 500°F at the combustion chamber to injector head flange (maximum soakback at flange)

MISSION OPERATIONS

The general functions of the reaction control engines on the service module are to provide translational and rotational capability for the Apollo spacecraft during the Lunar Apollo mission.

Upon receiving thrust control signals from the stabilization and control system, guidance and navigation system, or the rotational/translational hand controller, the reaction control engines provide the required impulse(s) for the accomplishment of the normal spacecraft maneuvers outlined below.

NOMINAL PULSE PERFORMANCE CHARACTERISTICS

PFRT CONFIGURATION-TYPE ENGINES

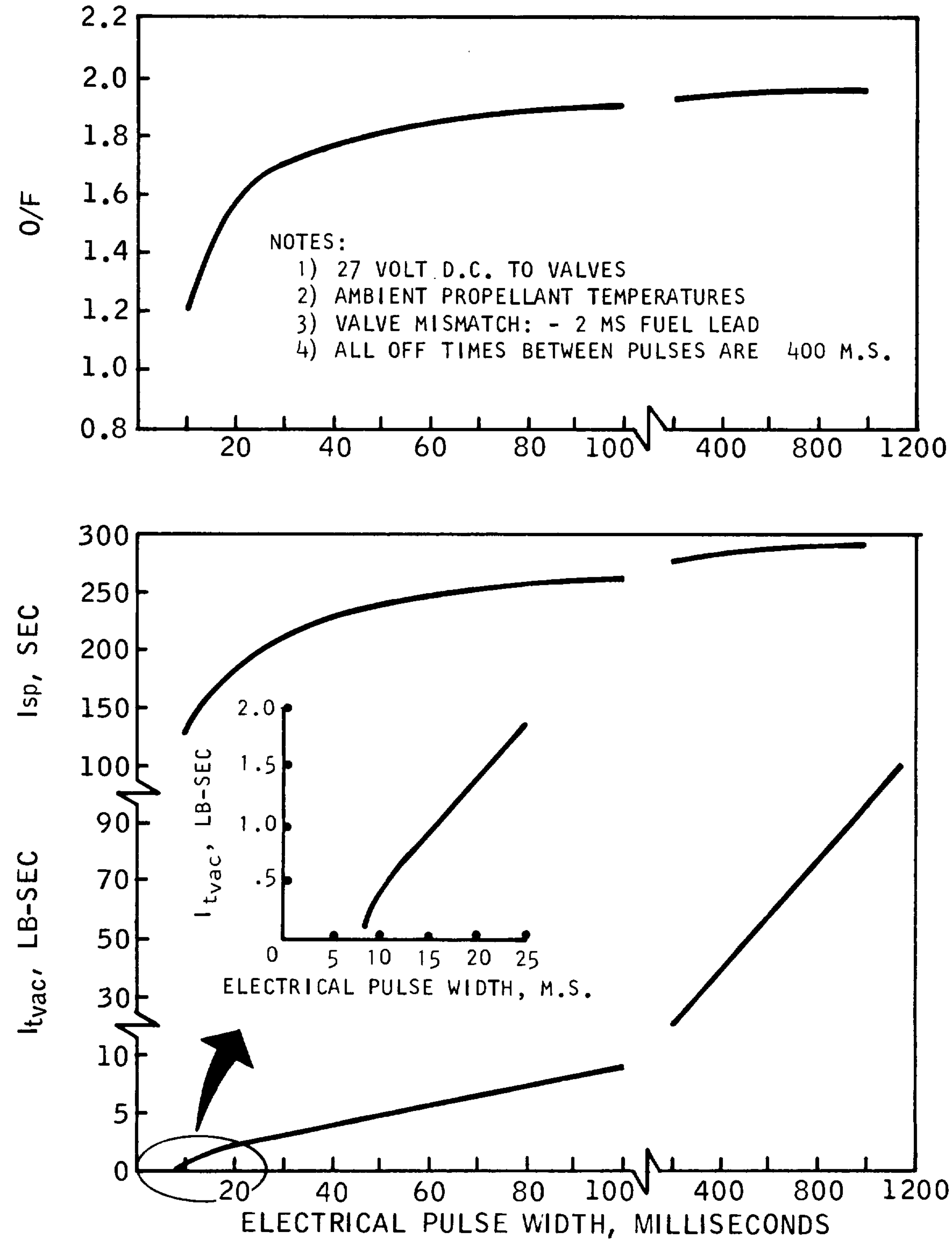


Figure 21

Spacecraft Separation from Boost Vehicle - Lunar Excursion Module

After injection of the spacecraft into a translunar trajectory, the service module reaction control system provides the velocity increment required for spacecraft separation from the SIVB boost vehicle-lunar excursion module.

Lunar Excursion Module Transportation and Docking

Upon completion of separation from the boost vehicle-lunar excursion module during the translunar trajectory, the service module reaction control system provides the velocity increments and/or the three-axis attitude control required for transposition and docking of the spacecraft to the lunar excursion module.

Spacecraft-Lunar Excursion Module Separation from Boost Vehicle

After transposition and docking of the spacecraft to the lunar excursion module during the translunar trajectory, the service module reaction control system provides the velocity increment required for spacecraft-lunar excursion module separation from the SIVB boost vehicle.

Stabilization and Control

During a lunar mission, the service module reaction control system provides the thrust vectors required for three-axis stabilization and attitude control of the spacecraft in all flight phases after translunar injection, and prior to command module-service module separation. These flight phases include translunar and transearth trajectories and lunar orbits.

Service-Propulsion System Maneuvers

The service module reaction control system is used in conjunction with the service propulsion system for those maneuvers in which the service-propulsion system is required for major changes in velocity. The service module reaction control system provides the initial positive acceleration force required for settling of service module propulsion system propellants prior to a zero g service module propulsion system engine start. In addition, the service module reaction control system provides the thrust vectors required for spacecraft roll control during periods in which the service module propulsion system is active.

Orientation Maneuvers

The service module reaction control engines provide the thrust vectors necessary for three-axis orientation maneuvers of the spacecraft prior to such critical events as midcourse ΔV maneuvers, navigational sightings, lunar orbit acquisition, inertial measurement unit alignment, and transearth injection.

Navigational Sightings

The service module reaction control engines provide the minimum impulse control required to obtain accurate navigational sightings.

Command-Service Module Separation from the Lunar Excursion Module

The necessary velocity increment required for command-service module separation from the lunar excursion module is provided by the service module reaction control engines.

Lunar Orbital Maneuvers

If required, the service module reaction control system provides the minor velocity increments necessary for spacecraft rendezvous with the lunar excursion module.

Command-Service Module Separation

The service module reaction control system provides the velocity increment required for separation of the command module and service module prior to the re-entry mode.

Post Atmospheric Abort

Prior to activation of the service module propulsion system in a post atmospheric abort mode, the service module reaction control system provides the emergency velocity increment required for spacecraft separation from the boost vehicle.

Earth Orbit Emergency Retrograde

Under normal conditions, the service module propulsion system provides the spacecraft retrograde velocity required for entry of the command module from the earth orbital mode. In the event of a service module propulsion system malfunction, the service module reaction control system provides the retrograde velocity (in extreme emergency) required for the de-orbit maneuver.

Operating Modes

The engine is capable of continuous operation, pulse modulated operation, or any combination of these modes. The maximum duration required for continuous operation during the Lunar Apollo mission is 500 seconds. This requirement is for the emergency retrograde maneuver during earth orbit. In accord with the requirements of the Lunar Apollo procurement specification the engine is capable of pulse mode operation under the following conditions:

1. Extended pulse mode operation at frequencies up to 25 cycles per second.

2. Delivering minimum impulses at intervals of 500 milliseconds.

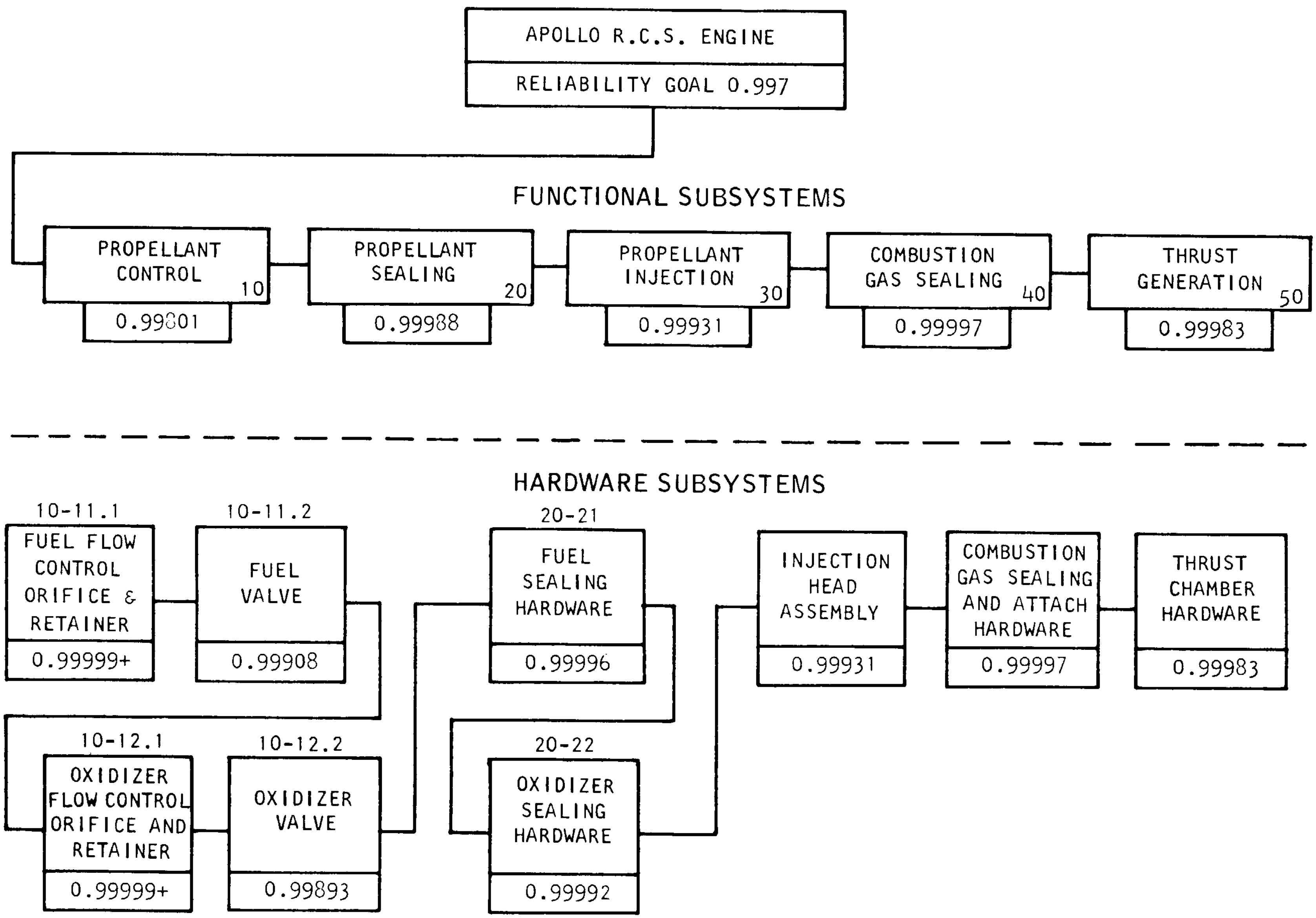
3. Restarting after receiving an electrical restart signal not sooner than 10 milliseconds after receiving an electrical shut-down signal.

The present engine has demonstrated the capability of operating beyond the above Lunar Apollo requirements. However, if these requirements were to be extended, additional documentation testing would be required. Insufficient testing was conducted in these extended areas during the Lunar Apollo program to verify performance, safety, etc. with any high degree of confidence.

RELIABILITY

The logic block diagram and the allocated component reliabilities for the PFRT engine are presented in Figure 22.

RELIABILITY LOGIC BLOCK DIAGRAM



This page intentionally left blank

III TECHNICAL ANALYSIS

ANALYSIS OF THE PRESENT ENGINE'S CAPABILITY FOR THE APOLLO X MISSION

The analysis conducted during this study program was based on the following "two-fold" approach.

1. The individual components were analyzed in detail to determine their capability for the Apollo X mission.

2. Also major functional and environmental areas of potential concern (i.e., meteoroid hazard, corrosion, etc.) were subjected to additional analysis.

The basic purposes of this "two-fold" approach were: (1) to act as a cross-check to insure that no potential problem areas were being inadvertently over-looked, and (2) to subject areas that were not included as specification requirements during the Lunar Apollo program to a much more careful scrutiny.

The basic objectives of this capability study were to determine the life limiting components of the SM/RCS engines and to define the environments, system requirements and/or mission requirements which were the limiting factors. Based on this information, the recommendations and development planning factor information was then formulated.

Because of the strong inter-relationship which exists between the engines, the system (both propellant and control), and the mission profile, an analysis was also conducted between these three to define potential trade-off benefits that might be beneficial from an engine operating standpoint. These potential trade-offs are mentioned as suggestions.

SOLENOID VALVE COMPONENT STUDIES

A detailed review of the present Apollo solenoid valves indicates that these valves will probably be suitable for the Apollo X mission. However, the information presently available in certain areas is insufficient to state conclusively that the solenoid valves are completely satisfactory within the reliability requirements of the Apollo X mission. Valve soak and cycle data with propellants is insufficient and actual test data on the valves at better than 10^{-9} mm Hg vacuum does not exist. A number of questions relative to propellant exposure and "hard" vacuum remain unanswered for the Apollo X mission. These are:

1. Will growth of the seat material into the propellant flow passage (which is the only place it can go) be sufficient so as to cause higher pressure drops due to either a decrease in stroke and/or

a change in the flow passage discharge coefficient? (It should be noted that at the Apollo N_2O_4 flow rates a change in stroke by 0.0004" results in a change of valve Δ P equal to 1 psi.

2. What are leakage rates of the valves with propellant, during and following exposure to "hard" vacuum (10^{-9} mm Hg) for prolonged periods of time, after occasional cycling, and at elevated temperatures?

3. Is there a significant lag in response when the valve is first actuated after having been exposed to propellants for a prolonged period of time?

4. Additional testing in these and other areas will be required to verify the acceptability of the valve for the Apollo X mission requirements. The following discussions present the rationale and analysis which is the basis for the above conclusions.

<u>Hard Vacuum (Less than 10^{-9} mm Hg)</u>

There are two phenomena which result from hard vacuum and which are of concern to proper valve operation. These are sublimation and cold welding.

Cold Welding

The only exposed surfaces which could present a problem due to cold welding is the metal to metal, poppet to seat interface. The materials in contact here are the type 446 stainless steel of the poppet and the AM-355 stainless steel of the seat. While these materials are dissimilar, they are sufficiently alike to pose the question of possible cold welding.

In general, cold welding is more likely to occur with surfaces which are completely clean. In the case of the valve, the surfaces are always subjected to propellant flow when separated. Consequently, it is expected that a thin propellant film will exist between the two surfaces even when the valve is closed and the surfaces are together. Whether or not this film will remain and for how long is difficult to predict.

Past test experience with the valves has been limited to an ambient pressure level of 7.5×10^{-6} mm Hg for leakage checks at 100 psia GN_2 inlet pressure. No adverse effects were noted. However, whatever "anti-friction" films have existed at 10^{-5} mm Hg are certain to be less significant at lower pressures. Since the materials are slightly dissimilar and it is likely that at least molecular films exist between the poppet and seat surfaces, it is expected that the valve will function satisfactorily. The only way to prove satisfactory operation is to actually subject the valve to pressure of 10^{-8} or 10^{-9} mm Hg and to actuate it under these conditions. It might also be mentioned that the actuator will develop a force of approximately 28 pounds where as only about 10 pounds are required to open the valve. The valve response characteristics could still be affected significantly if cold welding of any amount occurs.

Sublimation

The metals used in the construction of the valves that are continuously exposed to hard vacuum are all stainless steel with extremely low sublimation rates. No problems will occur. The other readily exposed material is the teflon used in the cable coating and in the seat assembly. Teflon also exhibits low sublimation rate and any material loss will be less than 1 percent during the 45 day mission. No problems are anticipated with the teflon seat.

Although the valve cover is installed by press-fitting, the cable egress through the cover is not intended to be pressure tight. Consequently, consideration should be given to sublimation of the potting compounds (Minnesota Mining and Manufacturing compounds No. 248 and 250), and the coil wire insulation (DuPont's Pyre-ML). The sublimation characteristics of these materials are not presently available.

Propellant Exposure

To date, the Apollo solenoid valves have been exposed to propellants continuously for periods of 21 and 30 days. These propellant tests were not conducted on the latest Apollo valve configuration; however, the materials were similar. Two fuel and two oxidizer valves were subjected to 28,000 actuations each during the 21 day exposure test. There was no leakage outside of specification requirements as well as no change in performance requirements during the test program. The two fuel and two oxidizer valves subjected to the 30 day propellant exposure exhibited zero leakage after the test. Tear-down inspection of these valves indicated that they were in excellent condition with no evidence of corrosion. Unfortunately, no tests of longer exposure have been performed. Separate propellant soak tests of the seat material have been conducted and it was determined that with N_2O_4 the specimen weight and volume increased by about 4 percent over a period of 10 days; thereafter, they remained fairly constant.

Seat Leakage

One of the most important functions of the solenoid valves is to provide a positive seal so that no propellant leakage occurs throughout the mission. Propellant leakage not only results in loss of critical propellant, but could also alter the ignition characteristics of the engine such that a potential catastrophic failure mode could result.

Table VI is an engineering tabulation of some significant valve tests that were conducted under semi-documented conditions during the development phase of the Lunar Apollo program. No reliability analysis of this data has been conducted because it is not inclusive of all test data and also there are minor variations in the design configuration of the valves used in the various tests which negate the lumping of the data together. However, the data is valuable from an engineering judgement viewpoint. The little-or-no-change in the seat leakage rates throughout these tests indicates the design capability of the Marquardt soft-seat configuration.

An endurance test was recently completed on the final valve configuration for the Lunar Apollo mission. A fuel and oxidizer valve were subjected to 1,000,000 actuations each, with water as the test fluid. No leakage (either forward or reverse) was detected at any time across either valve poppet during periodic GN_2 leak tests conducted throughout the test program. Based on this test and on other tests on similar valve seat assemblies, it is anticipated that seat leakage will not be a major problem for the Apollo X mission; however, additional testing is required to document the valve characteristics in accord with the higher reliability requirements of the Apollo X mission.

Contamination

The solenoid valve is the component of the engine assembly which is most sensitive to contamination from the system or propellant. The most critical area is the valve seat where contamination could result in propellant leakage. The soft-seal concept utilized the SM/RCS valves has the advantage that it can absorb small quantities of particles without leakage occurring. Contamination in other critical areas could result in a change of the valve response characteristics by changing valve stroke or causing binding of the armature. A detail analysis of the overall aspects of the contamination problem is presented in subsequent sections. It is reasoned that the contamination problem of the solenoid valve is not significantly greater for Apollo X than for the Lunar Apollo since the total burn times and starts are not greatly different.

Contamination testing of the Apollo solenoid valve is presently in progress with water containing the following minimum particulate levels per 100 ml of water:

Particle size microns	Number of Particles
25-50	1960
50-75	110
75-100	12

TABLE VI

SUMMARY OF SIGNIFICANT VALVE TESTS

Valves	Date	Type of Test	Propellant Exposure Time	Engine Burn Time sec.	Number of Actuations	Fwd. Leakage, cc/hr Before	Fwd. Leakage, cc/hr After	Remarks
T-3190 (Oxidizer) (Fluoro-gold valve seat)	Nov. & Dec. 1962	Temp. Endurance Test 1)Heated to 225, 250, 275, 300 & 325°F for 1 hour each	- -	- -	- -	0	0	
		2)250,000 cycles	- -	- -	250,000	0	0	Valve disassembled and reassembled
		3)15-day prop. soak (N_2O_4)	15 days	- -	- -	0	+	
		4)Heated to 300 & 325°F for 1 hour each	- -	- -	- -	+	?	(+ indicates small leakage which stopped after several valve cycles)
		5)100,000 cycles	- -	- -	100,000	?	?	
		6)Heated to 325°F for 1 hour	- -	- -	- -	+	0	
X 19875 (Fuel) X 19876 (Oxidizer)	July & Aug. 1963	Prototype Test Program (Eng. X19900, S/N 005) 1) 4 mission duty cycles 2) Life Test (Longest run: 646 sec.)	?	4668 sec. 4668 sec.	21,892 21,892	0 0	4 cc 1.25 cc	Engine in excellent condition after test
225920 S/N 0004 (Fuel)	Feb. 1964	Development Tests (Eng. T-9290, S/N 0001-5) Longest on time: 3195 sec. 50,200 pulses: .015 on, .015 off	?	6393	59,356	Within MTS requirements	Seat Leakage within MTS tolerance	Leakage at seal between seat assembly and valve body
225930 S/N 0004 (Oxidizer)		7,869 Pulses: .010 on, .010 off 1,229 Pulses: .015 on, .025 off	?	6393	59,356		Seat leakage within MTS tolerance	Porous spot at one of wrench holes on seat assembly

TABLE VI (continued)

Valves	Date	Type of Test	Propellant Exposure Time	Engine Burn Time sec.	Number of Actuations	Fwd. Leakage, cc/hr Before	After	Remarks
2 Fuel Valves	Aug. 1964	LEM Magic Mt. Tests (Eng. X20666-1, S/N 1003 and 1004) Sea level test conditions (inlet press. was approx. 50 psia during nontest periods)	35 days	46 sec.	13	?	0	Valves in excellent condition after exposure.
2 Oxidizer Valves				46 sec.	13	?	0	
Prequal Config. X21427(F)	Sept. 1964	Endurance Tests Fluid = H_2O	- -	- -	1,000,000	0	0	Poppet showed slight signs of wear.
X21428(Ox) Similar to prequal valve			- -	- -	1,000,000	0	0	
Seat Assembly (T-3190) Similar but not identical	10-29 62	Endurance Tests (Fluid = H_2O)	- -	- -	1,000,631	0	0	
T-10707	6-9-64	Endurance Tests (Fluid = H_2O)	- -	- -	1,000,000	0	0	
T-10708 First fuel lead valves w/ series direct coil	6-9-64		- -	- -	1,000,000	0	0	

TABLE VI (continued)

Valves	Date	Type of Test	Propellant Exposure Time	Engine Burn Time sec.	Number of Actuations	Fwd. Leakage, cc/hr Before	Fwd. Leakage, cc/hr After	Remarks
		Propellant Exposure (per MTP-0002, App. H)						
225930 (Oxidizer)	Aug. 1963	Each day, valves were actuated as follows:	21 days (N_2O_4)	- - -	28,350	0	0	
225920 (Fuel)		Coil / Cycles / Press. (psia)	21 days (A-50)	- - -	28,350	0	0	
225996 (Oxidizer)		Auto 1,000 181 Direct 350 250	21 days (N_2O_4)	- - -	28,350	0	2.5	
225995 (Fuel)		Pulse Width / (cps) .015 2-4 .030 2-4	21 days (A-50)	- - -	20,640	0	0	
T-3190 (Fuel) (Flurogold Seat)	Dec. 1962 Jan. 1963	Temp. - Endurance Test 1) Heated to 275°F, 300 & 325 for 1 hour each	- - -	- - -	- -	0	0	
		2) 100,000 cycles	- - -	- - -	100,000	0	0	
		3) 15 day prop. exp. (A-50)	15 days	- - -	- -	0	0	
		4) 275°F for 1 hour	- - -	- - -	- -	0	0	
		5) 100,000 cycles	- - -	- - -	100,000	0	10 cc	Leakage = 0 after several valve cycles
		6) 300°F for 1 hour	- - -	- - -	- -	0	0	
		7) 325°F for 1 hour	- - -	- - -	- -	6 cc	0	Leakage = 0 after several cycles
		8) 100,000 cycles	- - -	- - -	100,000	0	0	

To date, 66,000 actuations have been performed with no seat leakage in excess of the specification limitation and no performance deteriorations during regularly performed checks. The valve has been disassembled several times to check for wear. The only unusual condition that has been noted is a few particles embedded in the seat.

Performance Characteristics

The present Apollo valves have been tested over required ranges of voltage, temperature, and pressure and satisfactory operation has been demonstrated. The response of the present Apollo valves during automatic coil operation is sufficient to achieve the minimum total impulse requirement of 0.4 ± 0.2 lb-sec within the voltage range and pulse width required. As the study requirements indicated no change in the performance requirements between the Lunar Apollo and Apollo X, the performance characteristics of the qualified valve should be more than adequate for Apollo X.

INJECTOR HEAD COMPONENT STUDIES

As long as the hydraulic characteristics of the engine assembly are not altered, no effect on the engine operational, ignition, thermal and performance characteristics should occur. This, of course, assumes that all input conditions are within specification and that the propellant system and the engine solenoid valves are operating satisfactorily. The only other factors that can affect the hydraulic characteristics of the engine assembly during the Apollo X mission are things such as contamination, corrosion, erosion, and leakage within the injector head. The injector head has been designed such that the desired hydraulic characteristics are obtained without the use of any moving parts, thus eliminating this major source of failure modes. Thus, only the more passive causes mentioned above are left to alter the hydraulic characteristics of the injector head assembly.

Contamination

In regard to contamination, the dimensions of the critical areas in the injector head are much larger than the largest particlulate size that would be found in the propellant system or propellant, provided that the present cleanliness requirements are rigidly adhered to. The smallest opening in the injector head are fuel film coolant holes directed at the preigniter exterior wall, which are 0.010 inch in diameter; whereas the largest size particle that should be present in the propellant system or propellant is 100 microns (0.0039 inch). There is also no known history of a contamination problem occurring in the injector head during the development program. Probably the most critical source of contamination for the injector head is the pre-launch and launch phase when the engine exit is open to the atmosphere. This is a potential problem for the upward facing engines where dirt may be rammed into the injector head during the launch phase. A detailed discussion on contamination is presented subsequently.

Corrosion

Corrosion in the injector head due to propellants exposure is not considered to be a major problem for the Apollo X mission. The most likely areas of corrosion would be that of a galvanic couple occurring in the areas where aluminum is in contact with stainless steel. The oxidizer, which is the most likely of the two propellants to cause corrosion, is in contact with parts made of only stainless steel. It should be noted that the time interval that the materials of the injector head are exposed to propellants is considerably less than the mission duration since the propellants remaining in the injector head dribble volume after an engine firing will vaporize off quite rapidly in the space environment.

Leakage

The possibility of any leakage occurring within the injector head or at the injector head to solenoid valve interface that would adversely affect the hydraulic characteristics of the injector head is very remote.

Analysis of the metal-to-metal joints within the injector head reveals that there is no possibility of leakage occurring in these areas for the Apollo X mission. The most probable area for leakage would be at the omniseals at the interface between the solenoid valve and the injector head. It should be noted, however, that redundant teflon omniseals are utilized at both the fuel and oxidizer valve. Data available indicates that long exposure time to a fixed load does not increase the percent cold flow of teflon significantly. The effect of high vacuum (10^{-10} mm Hg) and particularly combined environments (radiation, propellant exposure, hard vacuum, etc.) on the properties of teflon are not well documented. However, based on the present available information and the known differences between the Lunar Apollo and the Apollo X missions, no major problem is anticipated in this area. Additional testing will be required to document the characteristics under the longer exposure to combined environments.

Erosion

No erosion problems are anticipated on the qualified engine design. The only area where erosion could possibly occur is between the main doublets. This phenomena was experienced early in the development phases of the program on experimental engines; however, no evidence of it has been noticed on the preigniter engine design that has been subjected to numerous starts. This erosion phenomena results in the exits of the main doublet injector holes in aluminum material being slightly distorted and burred. It is apparently due to a small hydraulic "reverse" resultant "heel" that is formed in addition to the "forward" resultant when two liquid streams are impinged together. In this case, the "heel" is made up of hypergolid liquids and hence, localized pressures and temperatures may be created which could cause this type of damage. The effect of these changes in the doublet hole exit is to cause distortion of the propellant streams with resultant affect on the performance and ignition characteristics of the engine. The possibility of any such phenomena occurring at the preigniter doublet is negligible as the preigniter injector is made of stainless steel. The qualification program will provide additional documentation and if a problem is apparent, it is assumed that corrective action would be accomplished in the Lunar Apollo program.

Structural and Life Characteristics

Life capability of the preigniter chamber far exceeds the requirements of the Apollo X mission. Both analysis and test results indicate that the maximum temperature the preigniter chamber will experience is approximately 1000°F. This is much lower than the critical temperature of the CRES-A-286 material from which it is fabricated.

The preigniter chamber, preigniter tubes, and interior passages within the injector head are also designed to withstand an ignition explosion without any failure occurring. This is an additional design feature to make the engine as insensitive as possible to any circumstances, i.e., erratic valve operation due to out-of-specification voltages, that could possible occur during the mission, no matter how remote the possibilities.

Ignition Characteristics

One of the primary functions of the injector head is to inject and mix the propellants in the combustion chamber such that a smooth ignition will result with a minimum over-pressure. Because of the above requirement, and the complex and relative unknown characteristics of hypergolic ignition, TMC conducted a comprehensive program to develop a consistent preigniter concept. The present PFRT engine incorporates the preigniter design that has been proven during the development and documentation test phases.

The success of this design is best shown by the following pertinent information. Review of the large amount of statistical data obtained during these programs shows that for a fuel lead of 0-7 ms, less than 0.3% of the ignitions were higher than 250 psia and no ignition pressures above 450 psia have been encountered.

COMBUSTION CHAMBER COMPONENT STUDIES

As the boost and launch phases were assumed to be the same for both Lunar Apollo and Apollo X, this study dealt only with the effects of space operation on the capability of the combustion chamber for the Apollo X mission.

The failure modes on the combustion chamber can be subdivided into the following four major areas:

1. Meteoroid Hazard

2. Coating Life

3. Overpressure Ignition

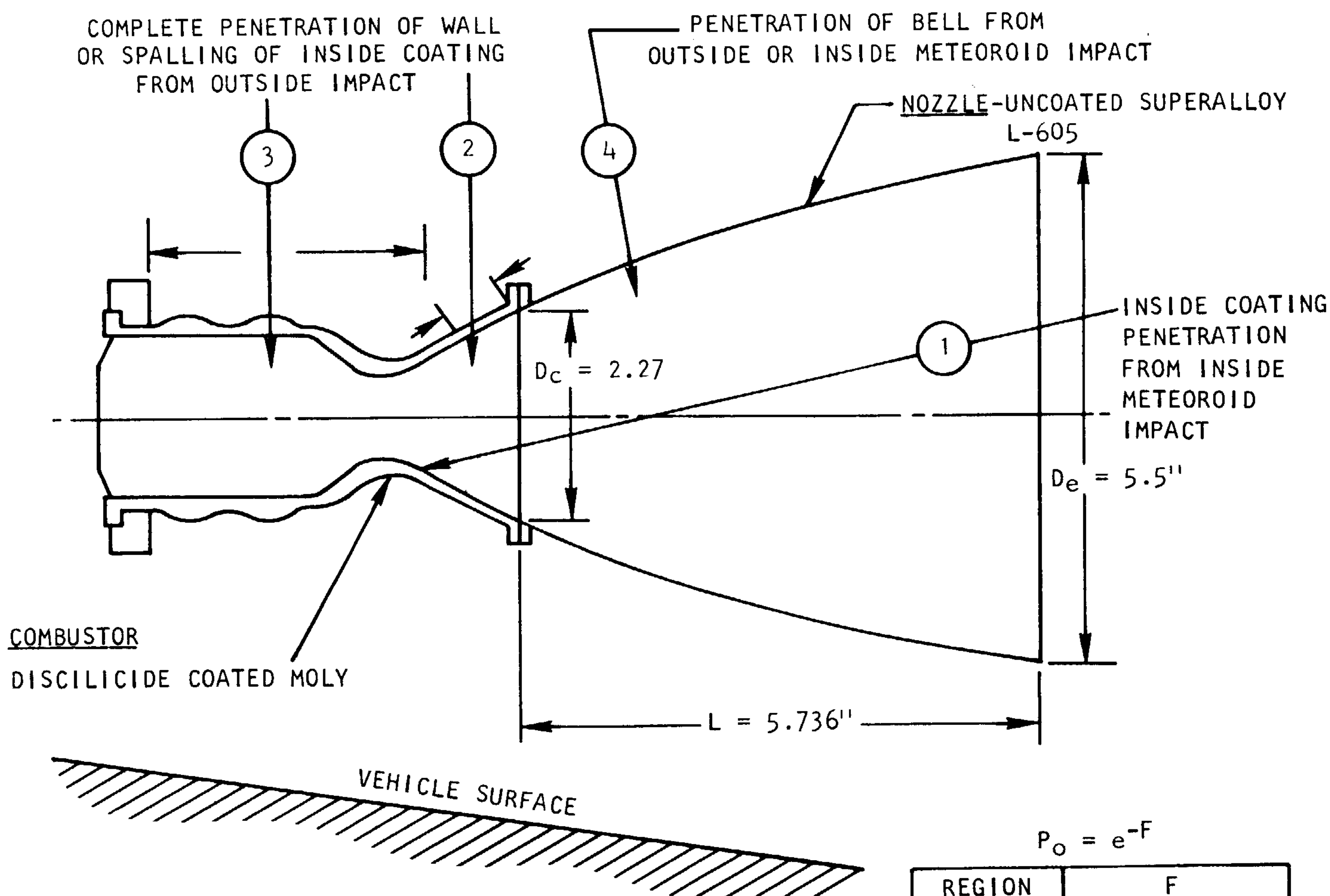
4. Fatigue, Cycling, and Dynamic Environments

Meteoroid Hazard

Figure 23 presents the likely modes of meteoroid hazard to the combustion chamber. Also included in the figure are the probability of no complete penetration or spalling for mission times of 34 and 45 days. The most sensitive area of the chamber to meteoroids is that of penetration of the inside coating downstream of the throat due to meteoroids entering through the window of the bell exit. For a 45 day mission, the probability of no penetration of this inside coating is 0.980. The overall probability of no penetration or spalling of the disilicide coated molybdenum combustor occurring is 0.9799. Hence, the probability of no combustion chamber damage is essentially equal to the probability of no inside coating penetration from inside meteoroid impact. Figure 24 is a plot of the probability of no complete penetration of the inside coating of the combustor versus mission time. The relative detrimental effect of spalling, complete coating penetration, or incomplete coating or chamber penetration on the life characteristics of the engine have not been documented.

The overall probability of no penetration of the L-605 bell from either an inside or outside meteroid impact is 0.939. However, penetration of the L-605 nozzle should not result in any significant reduction in either performance or life characteristics of the engine assembly; however, this assumption needs to be documented. If this does not constitute a thrust chamber failure, then the only possible problem in the bell area is that of possible flame impingement on the spacecraft through meteoroid impact holes.

ENGINE METEOROID HAZARD PREDICTION
APOLLO X MISSION



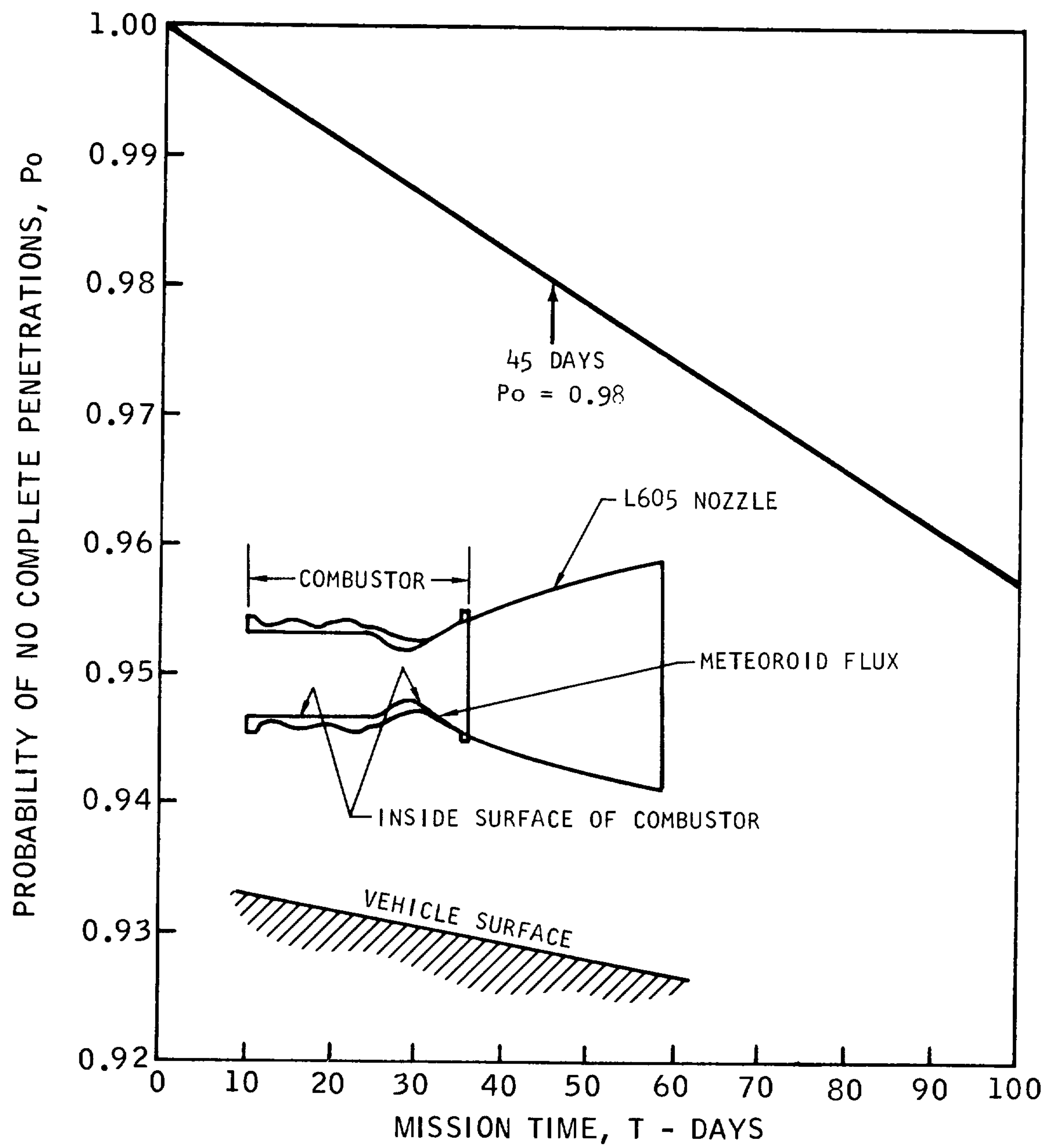
MISSION TIME, T (DAYS)	P_o = PROBABILITY OF NO COMPLETE PENETRATION OR SPALLING					
	1	2	3	4	2 OR 3	1 OR 2 OR 3
34	0.9851	0.99994	0.999982	0.9524	0.999925	0.9850
45	0.980	0.99992	0.999976	0.939	0.99990	0.9799

$P_o = e^{-F}$

REGION	F
1	4.37×10^{-4} T
2	1.68×10^{-6} T
3	5.28×10^{-7} T
4	1.40×10^{-3} T

Figure 23

PROBABILITY OF NO COMPLETE PENETRATION OF THE INSIDE COATING OF THE COMBUSTOR VS MISSION TIME



Coating Life

A molybdenum disilicide coating is applied to the base molybdenum metal to protect it from the oxidizing combustion environment. Thus, the chamber life is directly related to the coating life as loss of the coating will result in rapid oxidation and thus failure of the base metal. The coating weight loss due to sublimation in a hard vacuum while the engine is not operating is negligible.

The coating sublimation which should occur on the outside surface of the combustor during engine operation should not significantly affect the engine life, in a hard vacuum. Even if the outside surface coating completely sublimates, the combustor wall temperature will not greatly increase. The maximum combustor wall temperature is 2350°F (Nominal = 2250°F). As a frame of reference, if the outside surface emittance decreased even to one-half of the value, the maximum wall temperature would be equal to or less than 2720°F. However, there is the possibility that the coating will deposit on the near-by thermal control surfaces of the spacecraft and thereby alter their thermal characteristics.

Coating wearout failure during continuous engine operation is not considered to be an area of concern for the Apollo X mission. Figure 25 is an extrapolated plot of the predicted life characteristics of the combustor based on engine tests conducted at Marquardt at altitude conditions (0.05 psia). For the present engine operating temperature (maximum) of 2350°F, the curve indicates a chamber life of over one day of continuous operation. Even if the extrapolation is slightly off, the margin is so great that no problems should be encountered during the Apollo X mission. This should also be biased by the fact that only a fraction of the approximately 800 seconds of engine operation for the mission will occur at steady state operating times.

There is one possible failure mode which could present a potential problem for the Apollo X mission. This failure mode is based on the assumption that after an engine firing, the chamber pressure decays much more rapidly than the chamber wall temperature. Theoretical calculations indicate that the chamber pressure will decay to space vacuum in less than five seconds. After the solenoid valves close, the temperature decay from a steady-state temperature to 1000° or lower occurs in the order of one minute. During the time the walls are relatively hot and the pressure is very low, a significant amount of the silicon may be lost from the coating due to sublimation. Subsequent exposure to combustion gases could degrade the coating. Repeated cycles of exposure to vacuum and combustion gases possibly could result in a usable life significantly less than that which results from coating oxidation on the inside surface during continuous engine operation. Unfortunately, only meager data is available on the disilicide coating characteristics pertinent to a multi-start radiation cooled engine operating in a space environment. Thus, a conclusive statement relative to this failure mode cannot be made with any degree of confidence until additional testing has been conducted.

PREDICTED COMBUSTOR VS MAXIMUM WALL TEMPERATURE

DISILICIDE COATED MOLYBDENUM

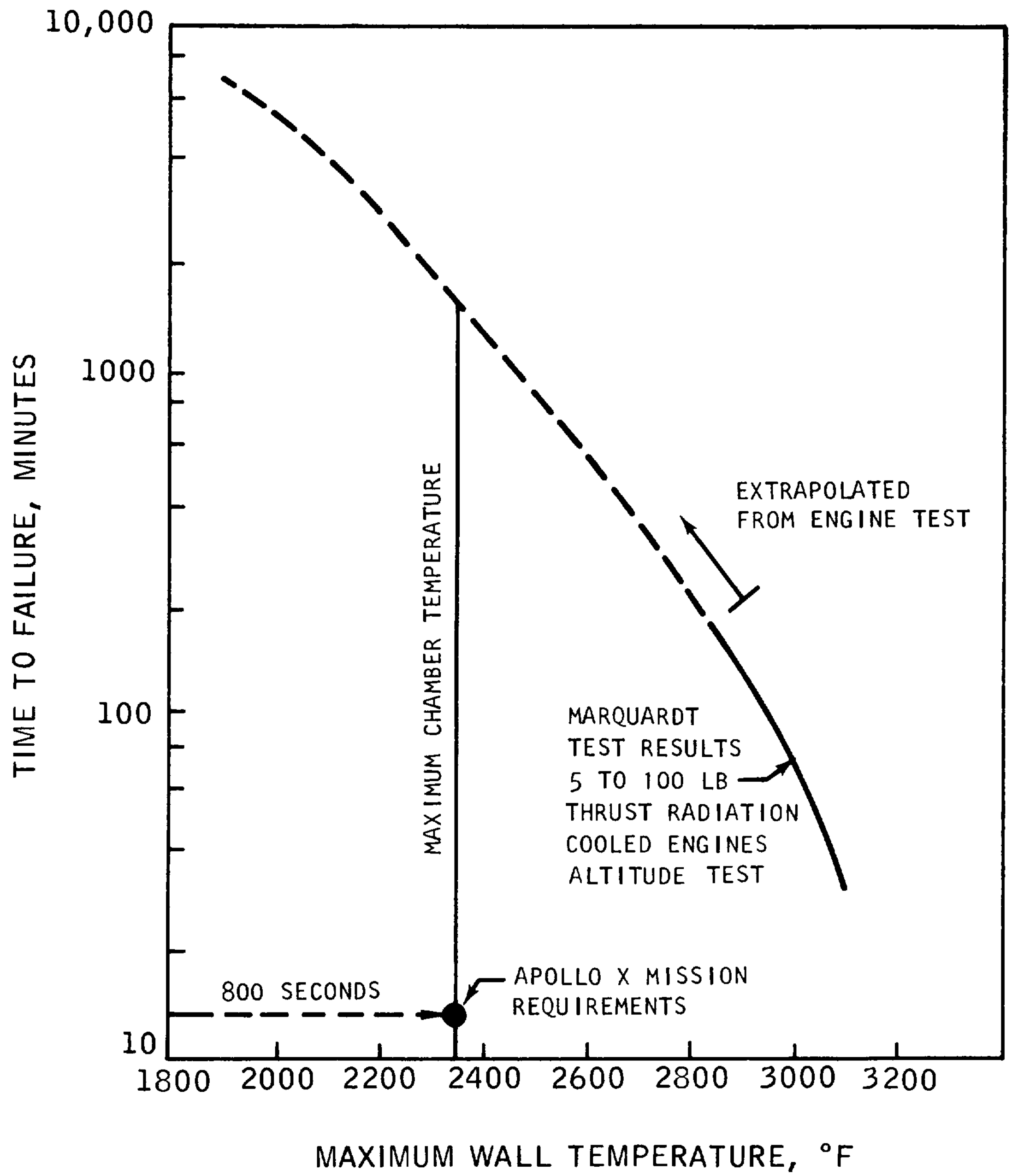


Figure 25

The effect of space radiation in combination with other space environments on coating life is probably not significant, but no data is available. Appendix A is a detailed presentation of the study conducted on the characteristics of the chamber coating.

Overpressure Ignition

The combustion chamber is protected by a three-fold approach to prevent a failure due to an overpressure ignition. First, a preigniter engine has been successfully developed which results in repeatable non-detonable ignition occurring at low ignition pressure levels over a wide range of operating conditions. Further details on the preigniter have been discussed in a previous section.

Second, the solenoid valves are designed to provide a fuel lead condition on both automatic or direct coil operation over the spectrum of specification conditions. The fuel lead provides an additional safety factor as previous testing conclusively proved that the ignition pressures under a fuel lead are statistically grouped closer together and are much lower than for an oxidizer lead.

Thirdly, the "beefed up" design of the combustion chamber with the thicker walls and circumferential ribs has been retained. A conservative structural analysis indicates the following design capability of the chamber that is subjected to uniform pressure loading at a surface temperature of -50°F.

Value with 50% factor of safety = 1720 psi

Overpressure burst = 1720 x 1.5 = 2580 psi

There is also a fourth factor that contributes to the evaluation of the overpressure ignition phenomena. This factor is the probability of an overpressure ignition of a large magnitude occurring even if none of the above design features were incorporated into the engine. This itself becomes very significant only when worst-case conditions such as low temperature propellant and chamber wall prevail.

Considering these design features, the probability of a failure due to an overpressure ignition occurring is considered to be extremely remote for the Apollo X mission even though more restarts will be required than for the Lunar Apollo mission.

Fatigue, Cycling, and Dynamic Environments

Fatigue failure of the chamber due to starting pulses is not considered to be a problem as a design criteria of 18,000 starts was utilized for Lunar Apollo and this is significantly higher than the approximately 10,270 starts anticipated for Apollo X.

The effect of space vibration on the structural integrity of the combustor or the bell joint is also not considered to be a problem for the Apollo X mission. The major thermal stresses that the Apollo X combustion chamber will experience will be due to the following causes:

1. Thermal shock due to engine start-up

2. Thermal cycling due to repeated cycles of cool down to the space environment from engine operating temperatures

3. Thermal cycling of a non-operating engine due to the combined effect of space environment, orbit profile, and vehicle attitude.

The design margins for the Lunar Apollo are sufficiently high that the slight increases in the Apollo X mission duty cycle requirements will have no appreciable effect on Items 1 and 2. The estimated magnitude of the thermal cycling resulting from Item 3 are significantly small from a temperature delta standpoint that no problems are anticipated.

OTHER COMPONENT STUDIES

The only other components not covered in the above discussions are: the seal between the injector head and the chamber, and the attach hardware.

A detailed analysis was not conducted on the chamber to injector head seal as a detailed development program is presently in process to develop a seal with higher thermal resistance for possible use on the passive thermal control engine. This is the engine which will be qualified for the Block II Lunar Apollo missions. The present L-605 seal in the PFRT engine design has presented no problems. The main chamber film cooling maintains the molybdenum flange at approximately 500°F (maximum soak back temperature) which is well below the critical temperature of the L-605 seal.

A structural review of the attach hardware indicated that there are no significant differences between the Lunar Apollo and Apollo X mission which would degrade the structural integrity or function of this component.

CORROSION STUDIES

The PFRT engine design was reviewed to determine the types of materials that are in direct contact with either N_2O_4 and/or Aerozine-50. The "Titan II" handbook (Reference No. 3) was utilized as the baseline for defining the compatibility of the above basic materials with the propellants. Table VII is a listing of the known static compatibility characteristics of

TABLE VII

STATIC COMPATIBILITY OF CURRENT APOLLO MATERIALS IN N_2O_4 AND AEROZINE-50

"A" Rating unless otherwise noted (* indicates "B" rating)

Part	Material	N_2O_4 Temp. °F	N_2O_4 Time (days)	Aerozine-50 Temp. °F	Aerozine-50 Time (days)
Housing (227997) Ring (228131-7)	6061-T6 (anodized)	60 130	180 30	60 160	270 90
Orifice (227423) Spring (227991) Retainer (227826)	304	60 100	180 7	60 160	270 90
Valve Plug (227824) Body Weldment (227817)	446	-	-	-	-
Armature (227825)	(446) Electrolize	-	-	60	360
Seat (228130) Standoff (228131-3) Orifice (228132) (228133)	A-286	60 100	180 7	60	270
Insert (227822) Seat (227917)	AM 355	65	90	160	180
Ring (5000-25-H)	PH 15-7 Mo	65	90	160	90
Spring (227827)	Inconel X	-	-	-	-
Weld Rod (227817)	92	-	-	-	-
Omniseal R 10205-012-A1N R 10205-021-A1N	Teflon (TFE) (over PH 15-7 Mo)	65 60	30 180*	60 160	270 30*
Seal (227921)	Teflon (FEP) (over 17-4 PH)	65 160 60	30 7 180*	60 60	180 270*
Seal (227821)	Teflon (TFE) glass filled	60	180	60	180

"A" Corrosion rate less than 1 mil per year for metals. Volume change 0 to +25%, hardness change $\pm$ 3 for nonmetals.

"B" Corrosion rate 1-5 mils per year for metals. Volume change -10 to +25%, hardness change $\pm$ 10 for nonmetals.

Reference: AFBSD-TR-62-2, "Titan II Storable Propellant Handbook," Revision B, March 1963, Bell Aerosystems Co.

these materials. Published data were not available in all cases for the mission duration or range of conditions that might be encountered on the Apollo X mission.

The area which has the greatest potential for being a problem is in the fuel passage where a stainless steel preigniter tube is in contact with the aluminum housing.

As extrapolation of chemical corrosion rates for either time or temperature effects is virtually impossible, additional testing should be conducted to determine the corrosion resistance capabilities of the engine, especially in areas where galvanic corrosion is possible.

STRUCTURAL STUDIES

A review of the structural design of the present Apollo engine was conducted from the viewpoint of the Apollo X mission requirements. The basic conclusion was that no significant differences in structural capability exist between the Lunar Apollo and Apollo X missions except for possibly the following two areas:

1. Sublimation of the chamber coating under the combined conditions of mission duty cycle operation and space environments.

2. Meteoroid hazard due to long-term space environment exposure.

Both of these areas are discussed in detail in a previous section.

The basic reason for the comparable structural capability between the Lunar Apollo and the Apollo X is that one-half hour (1800 seconds) material allowables and 18,000 starting pulses were utilized as the design criteria for the Lunar Apollo. The mission duty cycle (see Table II) estimated for Apollo X indicates that only 770 seconds of total accumulated burn time and 10,270 starts will be required for the 34-day Lunar polar orbit mapping mission. Thus, the only area of concern in regard to the structural capability is the possible effect of the longer exposure to space environments as indicated in the above exceptions.

One of the most severe segments of the mission in regard to structural requirements is the launch and boost phase. As the study requirements indicated no change in the launch and boost environments for the Apollo X mission from the Lunar Apollo mission, no problems should be experienced in this area. If the mission profile for the launch and boost phase is changed in the future, a restudy should be made of this area.

Vibration in space operation is not expected to be critical for Apollo X because 10 minutes exposure at the combined conditions of maximum temperature and vibration was used for the Lunar Apollo criteria. In addition, the margins of safety for the space vibration conditions for the Lunar Apollo are sufficiently high to permit a 50 percent increase in the number of cycles. Space vibration conditions only exist during operation of the Service Module Propulsion System and since the study requirements indicated that there is no basic change in the total time or starts of the Service Module Propulsion System for Apollo X, no problems are anticipated in this area.

METEOROID HAZARD STUDIES

The only part of the engine assembly that is exposed to meteoroids is the combustion chamber and exit bell.. A summary of the meteoroid hazard to the combustion chamber and exit bell is presented in a previous section. A detailed analysis is presented in Appendix B.

CONTAMINATION STUDIES

Because of the higher reliability requirements, increased length of the mission, and larger propellant quantities, the area of contamination was reviewed in detail to determine whether this would be a problem area for the Apollo X mission.

Susceptibility of Engine to Contaminants

Table VIII is a summary of the critical areas of the engine assembly that could be affected by contamination. Also included in the chart is the likelihood of a contamination problem occurring and the effect of such contamination on engine operation.

The most sensitive area of the engine to contamination is the valve seat where contamination could cause propellant leakage. This could result in serious consequences such as excessive loss of propellant and detrimental changes in engine ignition characteristics such that catastrophic failure could result. The soft seat, however, is advantageous from the standpoint that if small particles are trapped between the poppet and the seat, the soft seat can absorb them and possibly no leakage will result.

The next most sensitive areas are: (1) the valve stroke; and (2) the small clearance between the armature and the valve body.

The injector head should be relatively insensitive to any contamination originating from the system or influents as the critical flow areas are sufficiently larger than the contaminant size.* However, this is based on the

- - - - - - -

* Based on single hard particles. Clustered particles have not been considered.

TABLE VIII

SUMMARY OF POTENTIAL CONTAMINATION PROBLEM AREAS

Critical Areas	Likelihood of Contamination Problem	Effect of Contamination on Engine Operation
Inlet Orifice	Very Remote - hole size much larger than particulate levels allowed by specs.	Change O/F and $\dot{w}_p$ with resultant effect on engine performance, life, and ignition characteristics
Gap between Plug and Armature	Possible hard particle would prevent valve from opening fully	Change in O/F and $\dot{w}_p$ with resultant effect on engine performance, life, and ignition characteristics
Clearance between armature and valve body	Possible, however, no evidence has been seen to date. Clearance is within same range as particulate size	Slow valve response characteristics with resultant effect on ignition characteristics and pulse performance
Valve Seat	Most likely area for contamination problem in engine	Propellant leakage with resultant loss of propellant and possible effect on ignition characteristics
Valve Exit	Very remote - hole size much larger than particulate level allowed by specs.	Change in O/F and $\dot{w}_p$ with resultant effect on engine performance, life, and ignition characteristics
Main Doublets	Very remote due to contamination within system or influents as hole sizes are much larger than particulate level allowed by specs. Possibility of contamination problem occurring during prelaunch, launch, and space undefined	Change in O/F and $\dot{w}_p$ with resultant effect on engine performance and life
Preigniter Doublet		Change in ignition characteristics
Window Holes		Change in hydraulic characteristics with resultant effect on main chamber ignition and engine performance (especially short pulse)

TABLE VIII (Continued)

Critical Areas	Likelihood of Contamination Problem	Effect of Contamination on Engine Operation
Film Cooling Holes	Most likely area for a contamination problem within injector head. However, still very remote due to contaminants within systems or influents as hole sizes are larger than particulate level allowed by specs. Possibility of contamination problem occurring during prelaunch and launch is greatest	Change in cooling characteristics of main chamber with effect on life of main chamber and head to chamber seal
Other Fuel Passages in Injector Head	No problem - other areas would be affected first	--
Other Oxidizer Passages in Injector Head	No problem - other areas would be affected first	--

assumption that the system and influents will be maintained to the cleanliness standards that are presently specified. Also, there is the potential problem of contaminants entering the injector head during prelaunch and launch phases.

Effect of Long Duration Missions

The contamination inherent in the engine assembly and system as a result of manufacturing and assembly should be no worse for the extended missions than for the present Lunar Apollo mission in terms of type or number of particles per given surface area. However, the total number of particles may increase because of the larger surface areas required for the increased amount of propellants. It is anticipated that the levels of contamination may decrease as a result of experience gained in Lunar Apollo.

The contamination induced in the engine assembly and system as a result of acceptance testing, handling, ground checkout, etc. should be less, or at least no greater, for the extended mission than for the present Lunar Apollo mission. As a result of experience gained during the present Lunar Apollo missions, handling procedures, checkouts, etc. will undoubtedly be refined to eliminate problem areas.

One of the major areas where additional contamination will be increased for the extended missions is inherent in the propellants loaded aboard for the mission. It should be emphasized that this is an increase only in absolute number of particles because of the larger quantities of propellant required. The type or number of particles per given quantity of propellant should not increase.

Contamination entering the injector head during the countdown and launch sequence should be similar for both the Lunar Apollo and extended missions. If tests are not conducted during the Lunar Apollo program to determine the hazard of this type of contamination, it is recommended that they be accomplished on the Apollo X program.

Contamination generated in the system during the long duration missions is somewhat of an unknown. The probability of particles being generated from sliding parts is certainly increased. However, there is a minimum of components where moving parts are employed (i.e., engine valves and propellant isolation valves).

The number of actuations required for the propellant isolation valve will probably increase very little, if at all. For a normal mission, all of the actuations are accumulated on the ground prior to launch.

The engine solenoid valves will experience a small increase in actuations for the Apollo X mission (10,270 total for the Apollo X mission). However, the valve has been designed to reduce wear to a minimum. To date, there has been no evidence of excessive wear that would result in a contamination problem during flowing conditions. This is based on visual inspection of valves subjected to long endurance tests. The most severe case of wear occurs during dry operation. As dry operation occurs only during ground checkout, there should be no increase in number of dry actuations between the Lunar Apollo and the extended missions. Previous tests at TMC have indicated that there is no problem if dry actuations are limited to 5,000 or lower.

For system-generated contaminants, other than those originating in the engine solenoid valves, it is the system contractor's responsibility to define the degree of such contamination and limit its generation to that consistent with the cleanliness requirements.

It is assumed that the system contractor will supply propellant conforming to the military specification to the engine inlet throughout the mission. Thus, it is their area of concern to investigate the effects of long-term space storage on propellant characteristic changes which could result in potential contamination, corrosion, or performance changes to the engine.

Contamination entering the injector head during space operation is also an unknown. Again, it is assumed that analyses and test flights conducted during the Lunar Apollo program will determine the degree, if any, of this area being a problem.

THERMAL ENVIRONMENT STUDIES

It was originally intended that a detailed study would be made of the effect of the thermal environment and mission profile on the thermal characteristics of the engine assembly. However, because of the recent redirection to a semi-passive thermal control engine for the Block II Lunar Apollo mission, it was decided to discontinue this study. It was felt that the thermal data generated based on the PFRT engine design would not be applicable to the engine to be used for the Apollo X mission. The major differences in the semi-passive thermal control engine design is that it will include a thermal standoff of the fuel valve and possibly a higher thermal resistance seal between the injector head and the chamber.

The requirements of the thermal interface between the engine and vehicle is not completely defined. The present concept being proposed for Lunar Apollo Block II is to delete the water-glycol thermal control system and to install heaters around the individual solenoid valves to prevent the propellants from freezing. The mode of operation for the heaters (i.e., continuous, thermostat control, or shadow side on only, etc.) has not been finalized. The final size of the heaters will be dependent upon such factors as final engine configuration, safe operating temperature regimes for the engine (both minimum and maximum), mode of heater operation, etc.

When the semi-passive thermal control design of the engine and system are established, an addendum thermal study should be conducted to determine the thermal characteristics of the reaction control system and to evaluate any potential thermal problems that are peculiar to the Apollo X mission. Because of the quasi-static attitude of the spacecraft for long durations during lunar mapping, both overheating and underheating could be potential problems.

MATERIALS STUDIES

A detailed study was conducted to determine the suitability of the alloys, plastics, coatings, etc. for the Apollo X mission. Both individual and combined environment effects were considered. Careful review of the pertinent published technical data revealed that significant gaps exist in certain areas, particularly that of multiple environmental exposure. A summary of the materials and space exposure conditions are presented in Table V.

Solar Radiation Effects

Solar radiation plays a minor role in the degradation of properties, in that its effect on materials is limited to exposed surfaces only. The damage to metals or plastics, which are shielded by structures, or are contained within components, is negligible.

The effect of ultra-violet rays on bare plastics is similar to that produced by gamma ray exposure in an oxygen bearing environment. However, no plastics in the engine assembly are directly exposed to solar radiation.

The combustion chamber with the Durak B coating is the only engine component directly exposed to solar radiation, and this is not considered to be a significant factor for the relatively short durations of the Apollo X mission.

High Vacuum Effects

High vacuum in space can be detrimental to materials in many ways. Among these are: cold welding of metals; evaporation of metals and plastics, surface coating evaporation, accompanied by changes in surface emittance; the effect on fatigue (low cycle), and creep life of metals.

Cold Welding of Metals

Laboratory tests have indicated that, under high vacuum, the normal surface impurities and oxide coatings of many metals will disappear. Under high contact loads, surfaces of certain materials have been caused to bond.

The accepted design practice of utilizing dissimilar metals, hardnesses, and metallic coatings, will tend to minimize such potential space welding. Tests have been run on stainless steel, aluminum alloys, etc. for three-week intervals at vacuums of 10^{-7} mm of Hg for both similar and dissimilar alloy combinations without any indication of welding. The results of this study are shown in Table IX.

TABLE IX

COLD WELDING OF BARE METALS IN HIGH VACUUM

ALLOY	VACUUM	EXPOSURE TIME	RESULTS
347	1.8×10^{-7} torr	3 weeks	No welds
6061-T6	1.8×10^{-7} torr	3 weeks	No welds
Ti-6Al-4V	1.8×10^{-7} torr	3 weeks	No welds
Hk 31-H24	1.8×10^{-7} torr	3 weeks	No welds

NOTES: 1 - (100 pound constant load on each set of mating samples)

2 - No welds were obtained when the above alloys were tested in dissimilar combinations.

Reference: USAF Report RTD-TDR-63-1050, "The Explorer Program," February 1963

The only area of cold welding of concern for the Apollo X is the poppet to seat mating of the solenoid valve. The materials in contact here are the Type 446 stainless steel of the poppet and the AM 355 stainless steel of the seat. While these materials are dissimilar, they are sufficiently alike to pose the question of possible cold welding. Because of long exposure times and high vacuum, evaluation of the potential cold welding effect of this area will be required.

Evaporation Rates

Evaporation rates of the metals utilized in the SM/RCS engine present no problem for the Apollo X mission. In the case of plastics, coatings, and elastomers, high vacuum "evaporation" can produce significant mechanical and physical property changes. Decomposition in high vacuum takes place throughout material cross section, and is not merely a surface effect. Decomposition rates usually increase exponentially with temperature and linearly with time. Tests conducted on teflon TFE and teflon FEP at 10^{-7} torr and 212°F for 100 hours indicated a weight loss of only 0.04 percent and 0.08 percent respectively. Since decomposition rates vary basically linearly with time, weight losses should be less than 1 percent for teflon materials during the Apollo X mission. The sublimation characteristics of the potting compounds and the coil insulation for the solenoid valve are not presently available. Additional tests should be conducted to determine these characteristics for the Apollo X mission. Tests conducted on phenolics and glass fiber laminates at 10^{-7} mm Hg and room temperature for 42 days resulted in no dimensional change and a negligible weight loss (0.08 percent).

Effect on Combustion Chamber Surface Emittance

TMC has conducted a considerable amount of work on the effects of vacuum combined with high temperature on the Durak B coating of the combustion chamber. Negligible effects were noted on either the coating or external surface thermal total emittance for samples subjected to 3200°F at 10^{-3} torr for periods up to 30 minutes. Limited tests (two samples) were run at 2500°F and 4×10^{-5} torr. The total hemispherical emittance after approximately 24 minutes stabilized at a value of 0.54 (this is for a grit-blasted chamber). The results of this test are shown in Figure 26.

Effect on Fatigue and Creep Life of Alloys

Only a limited amount of experimental work has been accomplished in the area of high vacuum effect on the fatigue and creep life of the alloys. The fatigue life (low cycle) of 316 stainless definitely improves under vacuum conditions (see Figure 27). Creep life results for other alloys were variable, depending on the test temperature.

HEMISPHERICAL TOTAL EMITTANCE VS TIME

2500 °F ON MOLY COATED WITH $MoSi_2$ IN VACUUM AT $4X10^{-5}$ TORR

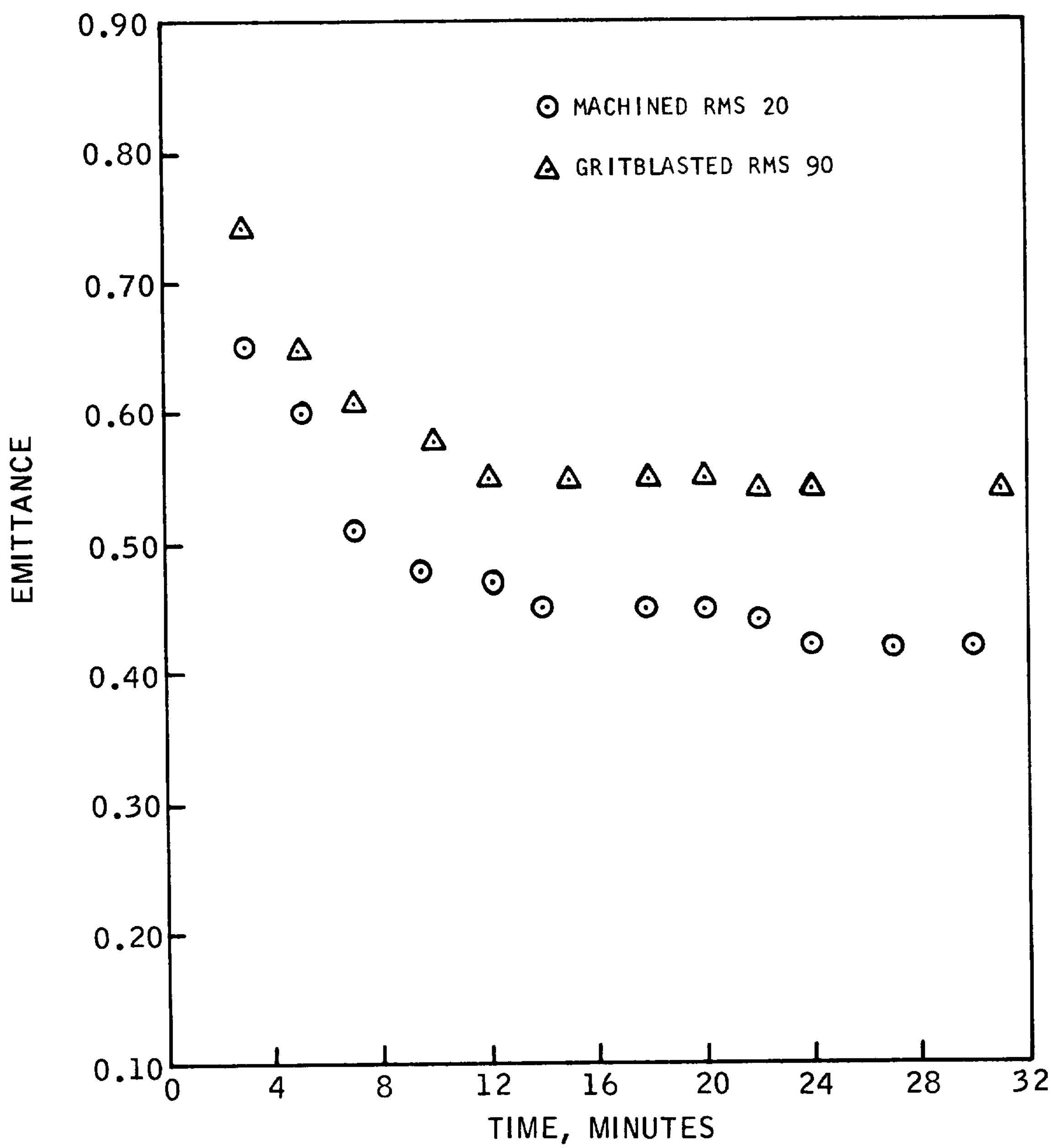
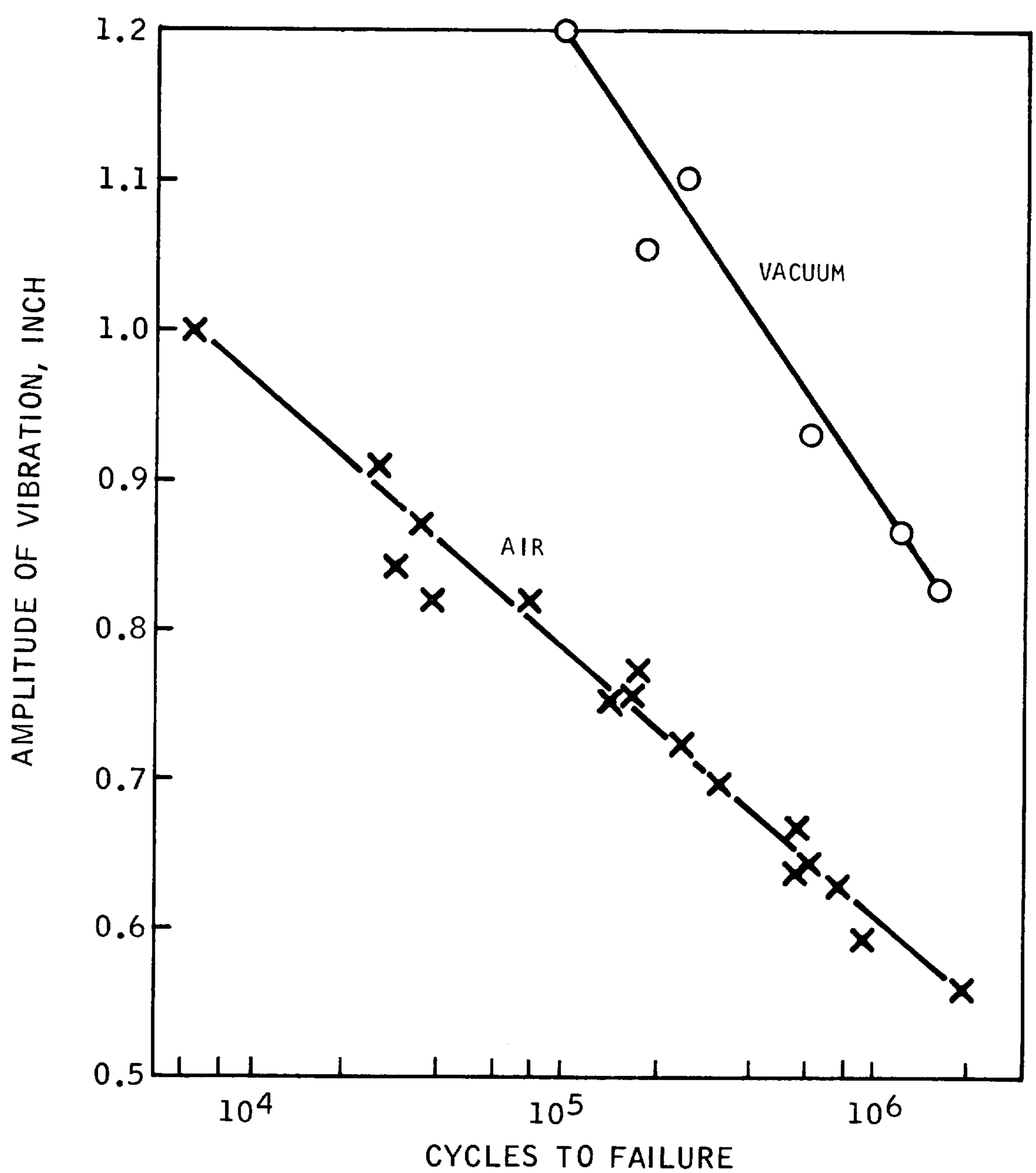


Figure 26

FATIGUE DATA FOR TYPE 316 STAINLESS STEEL

1500° F AND APPROXIMATELY 5 CPS



Compressive Cold Flow (Creep) of Teflon and Plastics

The amount of cold flow is very temperature sensitive; however, the percent of deformation levels off after only a short period of time for temperatures as high as 392°F. Figure 28 shows the effect of long-time compression on solid teflon as a function of both temperature and length of exposure. If the trend of this curve continues, there should be no significant effect on the cold flow properties of the teflon between the Apollo X and Lunar Apollo missions based on temperature and time exposure. The use of glass-filled teflon will further reduce the cold flow characteristics.

Multiple Environmental Effects on Materials

The simultaneous exposure of materials to combined environmental effects such as space radiation and vacuum, etc. is, in general, quite detrimental to many nonmetallic materials. Table V points out the various combined environmental variables to which current Apollo hardware will be exposed.

Significant data gaps exist in these areas of multiple environment exposure. Unfortunately, the possible interactions and total degradation of properties cannot be calculated or determined, in general, from existing data concerning individual environmental variables.

SPACE RADIATION STUDIES

Damage to materials by space radiation occurs by:

1. Ionization and excitation process (chief source of damage to plastics, insulators, and ceramics)

2. Atomic displacements (chief source of damage to metals, semi-conductors)

3. Transmutation (the changing of an element into another element - negligible in space)

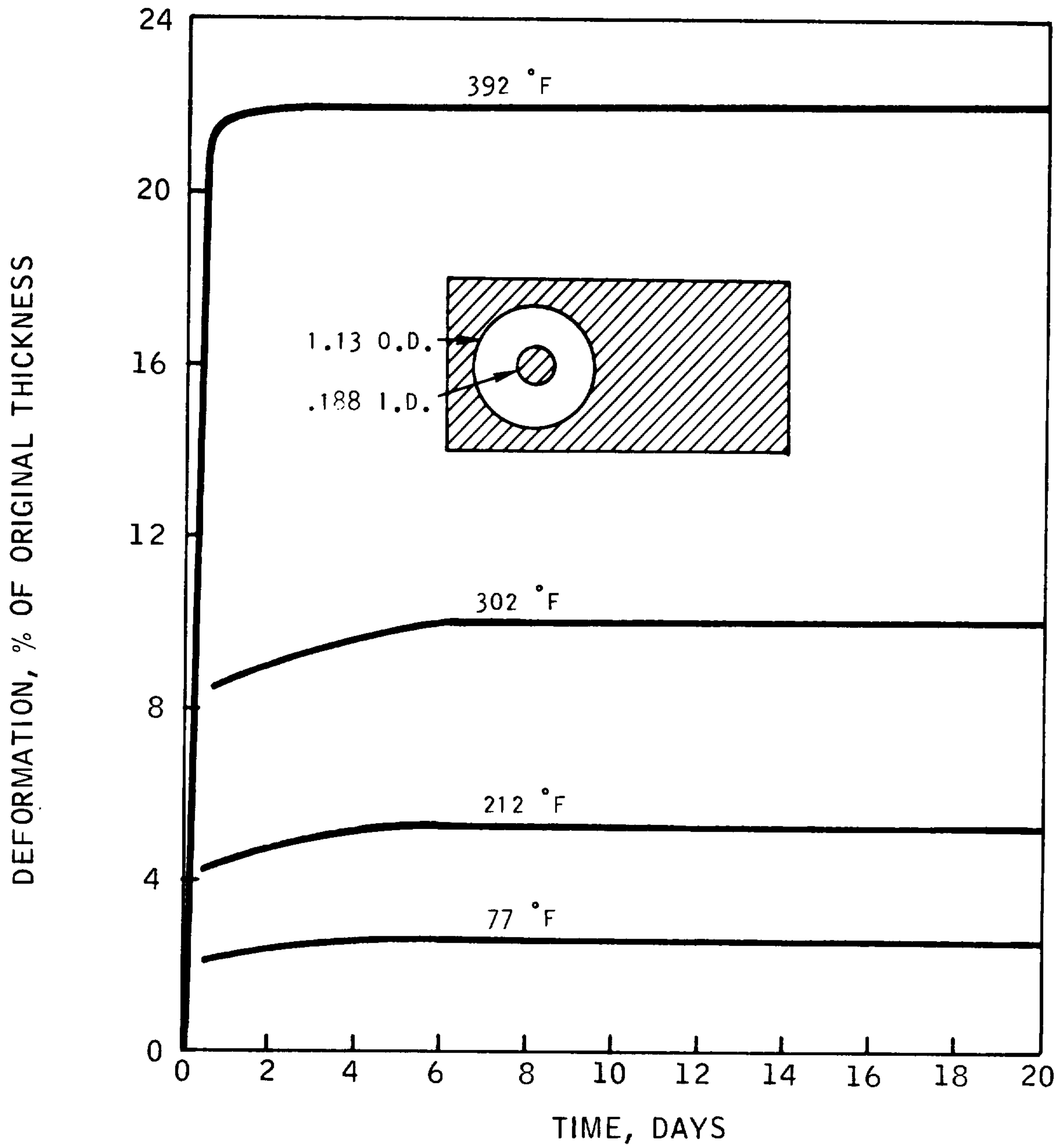
Electron damage can be stopped by thin shielding - approximately 0.05 cm of aluminum will block most of Van Allen Belt electrons. However, the absorption of electrons results in the generation of secondary electromagnetic radiation (brehmsstrahlung, or gamma rays) which then act on the materials behind the shielding. The energy levels of these secondary rays may equal those of the originating electron flux. The penetrating power of these gamma rays is much greater than that of the original electrons.

Proton effects, i.e., penetration for the same energy level as electrons is less severe. However, the upper energy level limit for protons may run as high as 10^3 Mev. (while electrons will only slightly exceed 1 Mev.) and thus can produce greater damage.

EFFECTS OF LONG - TIME COMPRESSION ON SOLID 'TEFLON'

1000 LB/SQ. IN.

REF:
"TEFLON FOR GASKETS AND PACKINGS", E.I. DUPONT DE NEMOURS, INC., / 1963



Specific Damage to Materials

Radiation damage to metals requires extremely high energy levels and is not of significance for the Apollo X mission. Radiation damage to plastics and other non-crystalline materials is almost independent of the type of energy received, but depends upon the total energy absorbed.

Of the Apollo X profiles, the most severe radiation will be encountered during a 45 day earth orbit at an altitude of 200 to 260 nautical miles and low inclinations.

The engine installation on the spacecraft is such that critical materials are shielded quite well. Figure 29 shows the typical structural shielding around the critical components of the engine assembly. In order for electrons or protons to reach the outlet side of the valve, they would have to travel more than 1 1/2 inches through the injector holes and up the propellant feed passages. Initial contact is then made with the metal poppet and metal-to-metal seat. It is believed that this path will not result in subjecting the critical materials, (i.e., valve seats, "O" rings, etc.) to more radiation than it experiences on the sides through the spacecraft structural shielding.

The design details of the structural shielding around the engine is undefined because of the passive thermal control evaluation that is presently being conducted. However, for this study program it has been assumed that it would be similar to that shown in Figure 29 and is constructed of 0.030 inch thick aluminum. Figure 30 shows the expected radiation dosage in rads that would result inside this structural shielding of engine mount installation for a 200 nautical mile earth orbit (30° inclination). From this curve is can be seen that the maximum expected radiation inside the structural shielding is approximately 1.5×10^4 rads for a 45 day mission.

The damage threshold radiation dosage for various materials is presented in Table X. If the anticipated radiation dosages of Figure 30 are compared with the susceptibility of the materials as defined in Table X, it is concluded that no detrimental damage should occur to the critical organic materials. The elastomers, potting, and insulation inside the solenoid valves are particularly well protected by the valve housing in addition to the structural shielding of this vehicle. The most sensitive material is teflon and its threshold dose is between $2\text{-}5 \times 10^5$ rads which should be sufficiently higher than the 1.5×10^4 rads that is anticipated.

GROUND HANDLING AND CHECKOUT

The ground handling and checkout philosophy has not been completely established for the Lunar Apollo missions and therefore this section is somewhat premature. However, basic observations will be made relative to the type of checkouts required to obtain the highest confidence levels in the engine prior to flight.

ENGINE MOUNT INSTALLATION

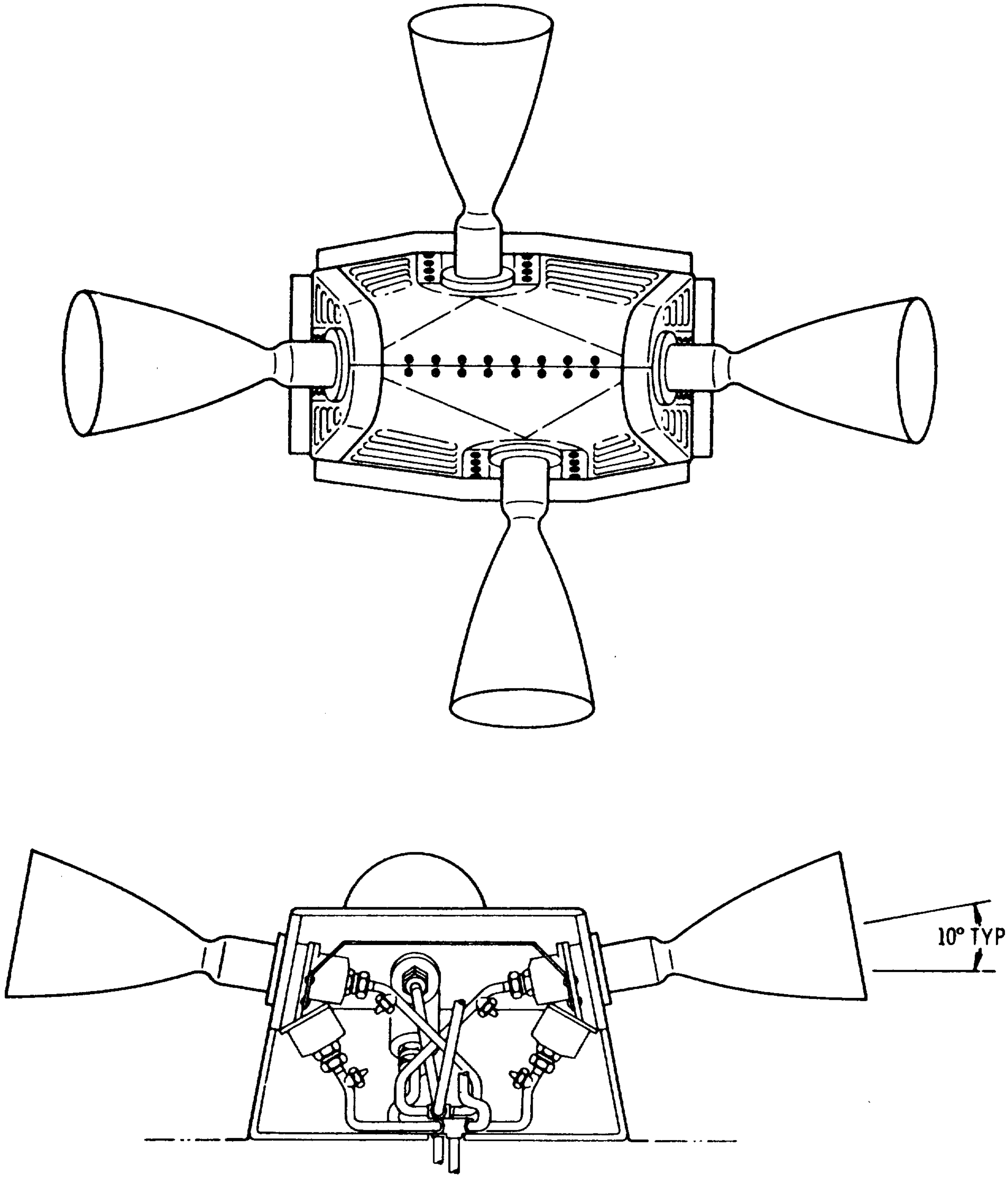


Figure 29

DOSAGE OF VAN ALLEN PROTONS AND ELECTRONS
DELIVERED TO ALUMINUM SHIELDING FOR A 200 NAUTICAL MILE EARTH ORBIT (30° INCLINATION)

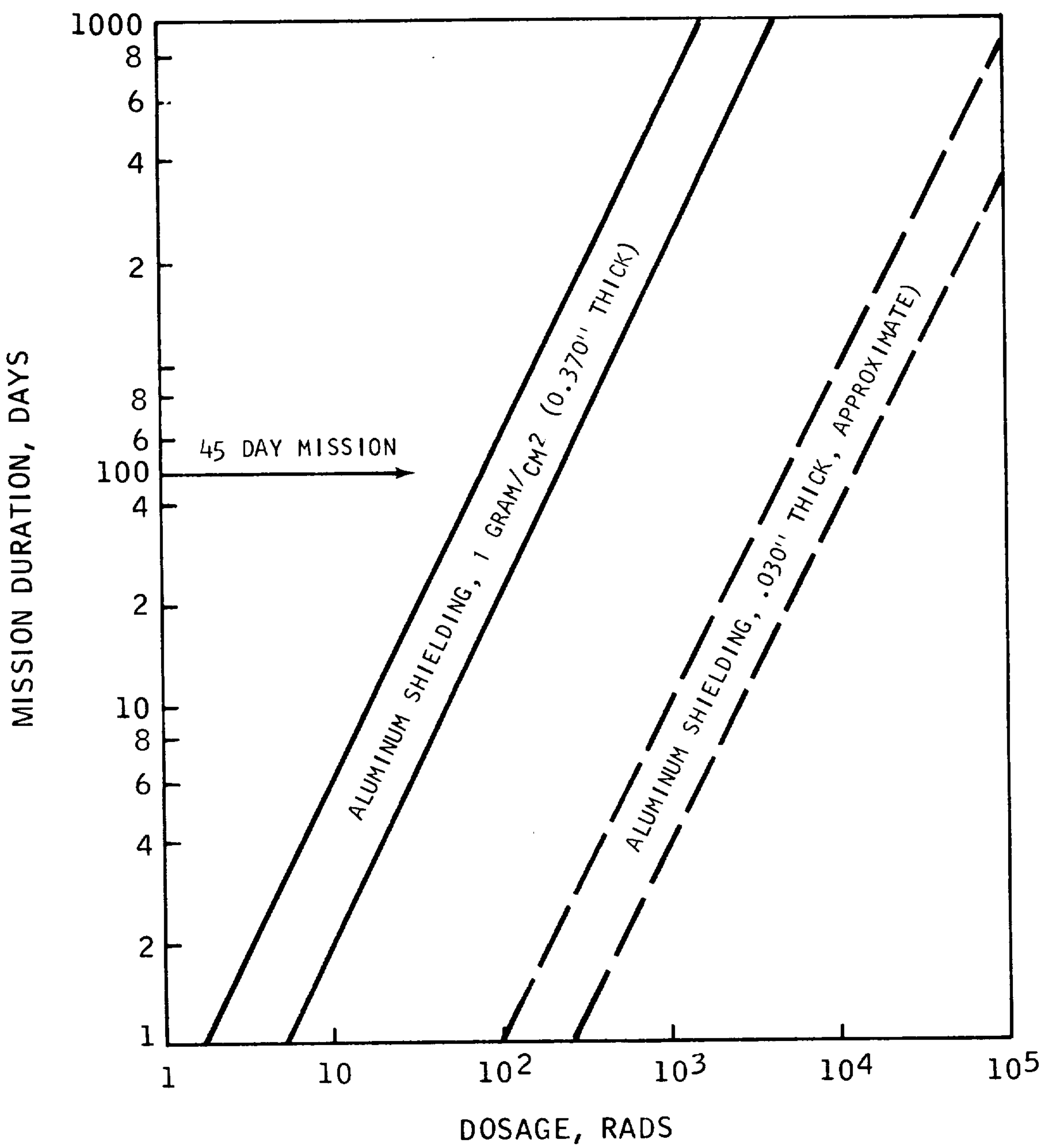


TABLE X

APPROXIMATE DAMAGE THRESHOLD (RADIATION) DOSE FOR MATERIALS

MATERIAL	GAMMA RAY THRESHOLD DOSE
	ergs/gm*
Glass	3×10^{7}
Teflon	$2 - 5 \times 10^{7}$
Nylon	$5 - 7 \times 10^{8}$
Graphite	5×10^{9}
Polyethylene	$1 - 2 \times 10^{10}$
Carbon Steels	5×10^{11}
Ceramic	$1.5 - 3 \times 10^{14}$
Stainless Steels	5×10^{13}
Carbon	$6 \times 10^{12} - 2 \times 10^{15}$
Copper	$7 \times 10^{13} - 5 \times 10^{16}$
Aluminum Alloys	$5 \times 10^{14} - 1 \times 10^{17}$

*Divide ergs/gm (c) by 100 to obtain rads

Reference: "Space Radiation and Its Effects on Materials," REIC Memo #21, Battelle Memorial Institute, 30 June 1961 (Also REIC #3, Page 9)

The procedure of static firing the engines during the checkout procedure at Cape Kennedy is felt to be entirely unnecessary and is in many ways, detrimental. Additional handling, fill, and drain procedures will be required where damage or contamination might be incurred by the engine assembly or components. Also sea level testing of an altitude combustion chamber is not conducive to maintenance of high reliability and confidence levels. As hypergolic propellants are utilized, there should be no concern about ignition occurring in flight. Valve response checks and flow calibration of the propellant system should be adequate to check the performance characteristics of the engine assembly.

It is considered that the highest confidence levels in the engine can be obtained by conducting flow and response checks in lieu of the static firing procedure.

PRE-LAUNCH AND LAUNCH PROTECTION

For the Lunar Apollo missions, the SM/RCS engines are exposed to the elements during the final hours on the pad (after the gantry is pulled away) as well as being subjected to the extreme environments, (i.e., aerodynamic heating, dynamic pressure, contamination, vibration, etc.) of the launch and boost phase. Even though the engines are not required to operate during these phases, various detrimental effects can occur which could adversely affect the operational and life characteristics of the engine for the subsequent parts of the mission.

For instance, critical contamination of the injector head of the engine, could occur as a result of airborne contaminants being rammed into engines on launch. Cursory evaluations should be conducted to determine methods of alleviating this potential problem area. Also weight savings on the engine could also be obtained if a unique method of pre-launch and launch protection could be developed which would delete the stringent structural design criteria for the engine due to the launch and boost environments as well as providing contamination protection.

MISSION DUTY CYCLE INTER-RELATIONSHIPS

The inter-relationship between the SM/RCS engines and the mission duty cycle, is an extremely complex subject which involves many systems, functional interfaces, and a multitude of potential trade-off considerations. There are any number of changes (i.e., lower minimum impulse bit, decreased angular rotation rates, etc.) which will result in fewer engine starts, lower propellant consumption, etc. However, each one of these will result in some compromise of technical, economic, or schedule considerations. Thus, the following discussions are meant to be nothing more than a cursory evaluation, of potential changes which would result in less severe demands on the SM/RCS engine. No attempt has been made to evaluate the overall mission affect of such changes.

Ground Operational Support System

The utilization of GOSS (Ground Operational Support System) in place of the on-board navigational sightings would significantly decrease the number of engine starts and total accumulated burn time for a given mission. If the crude approach is taken of deleting all navigational sighting orientations and navigational sightings from the 34 day Lunar Polar Orbit mapping mission, the following rough comparison is obtained:

	Total Starts	Total Burn Time
With Navigational Sightings and Orientations	10,270	770 seconds
Without Navigational Sightings and Orientations	5,874	612 seconds

Attitude Hold During Lunar Mapping

A cursory analysis was conducted on the operational trade-offs that exist for the roll engines during the lunar mapping phase of the mission where rigid attitude hold requirements prevail.

Figure 31 is a plot of the individual roll engine starts required for the mapping segment of the mission as a function of total impulse bit. Also shown on this plot are the effect of single engine vs coupled engine firing and the effect of equally distributing the engine starts among various numbers of quads. For an impulse bit of 0.6 lb sec per engine, the following comparison results:

Two Engines Coupled Operation

Two Quads (firings equally distributed)	1,060 starts per engine
Four Quads (firings equally distributed)	530 starts per engine

Single Engine Operation

One Quad	~ 530 starts per engine
Two Quads (firings equally distributed)	~ 264 starts per engine
Four Quads (firings equally distributed)	~ 132 starts per engine

STARTS VS IMPULSE BIT
ROLL ENGINES

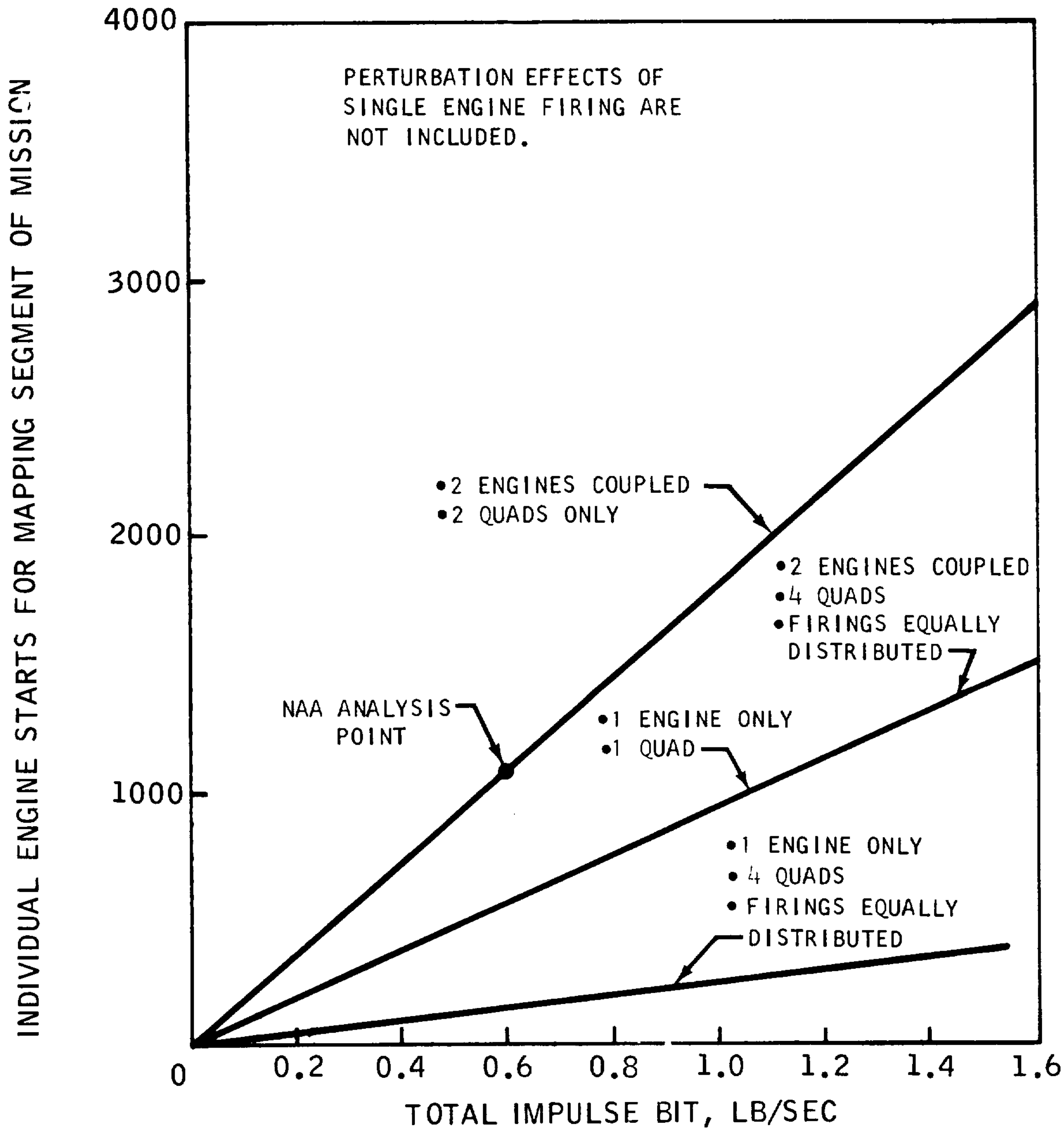


Figure 31

Single engine operation could be accomplished by installing inhibitors in the electronic control circuitry such as now exist on Lunar Apollo for "one shot" minimum impulse control during star sighting maneuvers. Lunar Apollo does not have this capability for limit-cycle operation however. The perturbation effects of single engine operation were not included in the cursory analysis as the necessary spacecraft information (weight, moment of inertia, etc.) was not available. A few additional starts may be required to damp out any such perturbations.

The Lunar Apollo spacecraft presently has the capability of distributing the roll engine starts equally among all four quads during two engines coupled operation. This is done simply by the astronaut selecting either the "A" or "B" roll systems. If this procedure is utilized, the number of starts for attitude hold during mapping would be decreased from 1,060 to 530 starts.

Further decreases in engine starts, burn time, and propellant consumption could be obtained by using a smaller minimum impulse bit. This could be accomplished by any number of methods. The present engine is capable of delivering minimum impulse bits consistently between 0.3 to 0.4 lb-sec for a given set of input conditions to the engine assembly. However, changes in voltage, temperature, and inlet pressure all effect the valve operating characteristics, thus the spread of minimum impulse bit over the range of specification conditions will vary. Probably the best method of consistently obtaining small impulse bits is to utilize a closed-loop system such as valve sensing circuit which will automatically command valve electrical off upon the oxidizer valve opening response being detected. With this sort of system, it could be possible to obtain impulse bits as low as 0.1 lb-sec with the pre-igniter engine configuration. Another alternative would be to utilize a valve (i.e., the bi-stable valve developed by TMC) which has faster response times and is relatively insensitive to voltage and temperature effects.

PERFORMANCE TRADE-OFFS

The performance capabilities of the present Lunar Apollo engine have not been optimized because of the combined trade-off between the schedule commitments and the performance required for successful completion of the Lunar Apollo mission.

Increases in both the steady-state and pulse performance characteristics could be obtained by optimizing the present engine design if technical and economical trade-offs and mission design indicate that this is desirable for the Apollo X mission.

SYSTEM TESTING

A very strong inter-relationship exists between the dynamic characteristics of the propellant supply system and the performance, ignition, and operating characteristics of the engine assembly. This is particularly true when several engines are fired simultaneously (either steady-state, pulse mode, or combinations thereof) from the same propellant system.

Detailed system testing at altitude conditions will be required to document the effect of this relationship. Just a few of the areas that require system documentation testing are:

Propellant Load Requirements

Propellant loads required for a given mission duty cycle may vary depending upon engine mission duty cycle. Even though the engine operates at a steady-state O/F = 2.0, the overall mission O/F may be significantly lower because of the hydraulic characteristics of the injector and systems at the short pulse widths.

Line Dynamic Characteristics

The effect of line dynamic characteristics during various modes of operation on the ignition characteristics of the engine requires further testing.

MISCELLANEOUS ITEMS

Other items that could have a possible inter-relation effect on the spacecraft are:

1. The possibility of the exterior combustion chamber coating sublimating and depositing on near-by thermal control surface, of the spacecraft. Such deposition of coating, if it occurs, could alter the critical thermal characteristics of the spacecraft.

2. A meteoroid hole in the bell section of the combustion chamber may result in possible flame impingement on the spacecraft.

The relative degree, if any, of the above items on the spacecraft capabilities are unknown. They are mentioned in this study program only as possible problem areas that the system contractor may not be aware of.

COMPONENT RELIABILITY ALLOCATION

The logic block diagram and the allocated component reliabilities for the Apollo X 34 day Lunar Polar Orbit mapping mission is presented in Figure 32.

FAILURE EFFECT ANALYSES

Table XI is a Failure Effect Analysis of the PFRT engine design for the Apollo X missions. Only possible failure modes resulting from the changes in the mission requirements and environments between the Lunar Apollo and the Apollo X mission were considered.

CONCLUSIONS

No major redesign of the engine assembly is anticipated for the Apollo X mission based on (1) the knowledge of the PFRT engine's operating characteristics, (2) the assumption that the Qualification Test will be successfully passed, and (3) the differences between the Apollo X mission and the present Lunar Apollo mission as enumerated in the previous sections.

There are however new requirements (i.e., meteoroids, more starts, increased reliability, etc.) being imposed upon the engine. In some cases, insufficient data is presently available to define the effect on the engine capabilities for the mission requirements of the Apollo X. As a result, a reasonably comprehensive test program will be required to verify the ability of the engine to meet the new requirements.

Additional engine documentation testing will be required to have a significant degree of confidence in the capability of this engine to perform for the Apollo X mission within the new reliability requirements. This is due to not only the increase in the reliability allocation to the SM/RCS engine for the Apollo X mission, but also because of the lack of reliability demonstration program for the present Lunar Apollo mission. Tests and analysis will also be required to evaluate the applicability of the Lunar Apollo Block II semi-passive thermal control design for the thermal environments and mission requirements of Apollo X.

RECOMMENDATIONS

It is recommended that the program plan outlined in Section IV herein be implemented, consistent with the proposed schedule. Any delay or reduction in program scope would result in a possible compromise in meeting the presently projected deliveries of hardware of the desired capability and reliability.

It is also recommended that an additional study program be authorized to evaluate the applicability of the Lunar Apollo Block II semi-passive thermal control design and concept for Apollo X missions.

LOGIC BLOCK DIAGRAM
ALLOCATED COMPONENT RELIABILITIES FOR APOLLO X

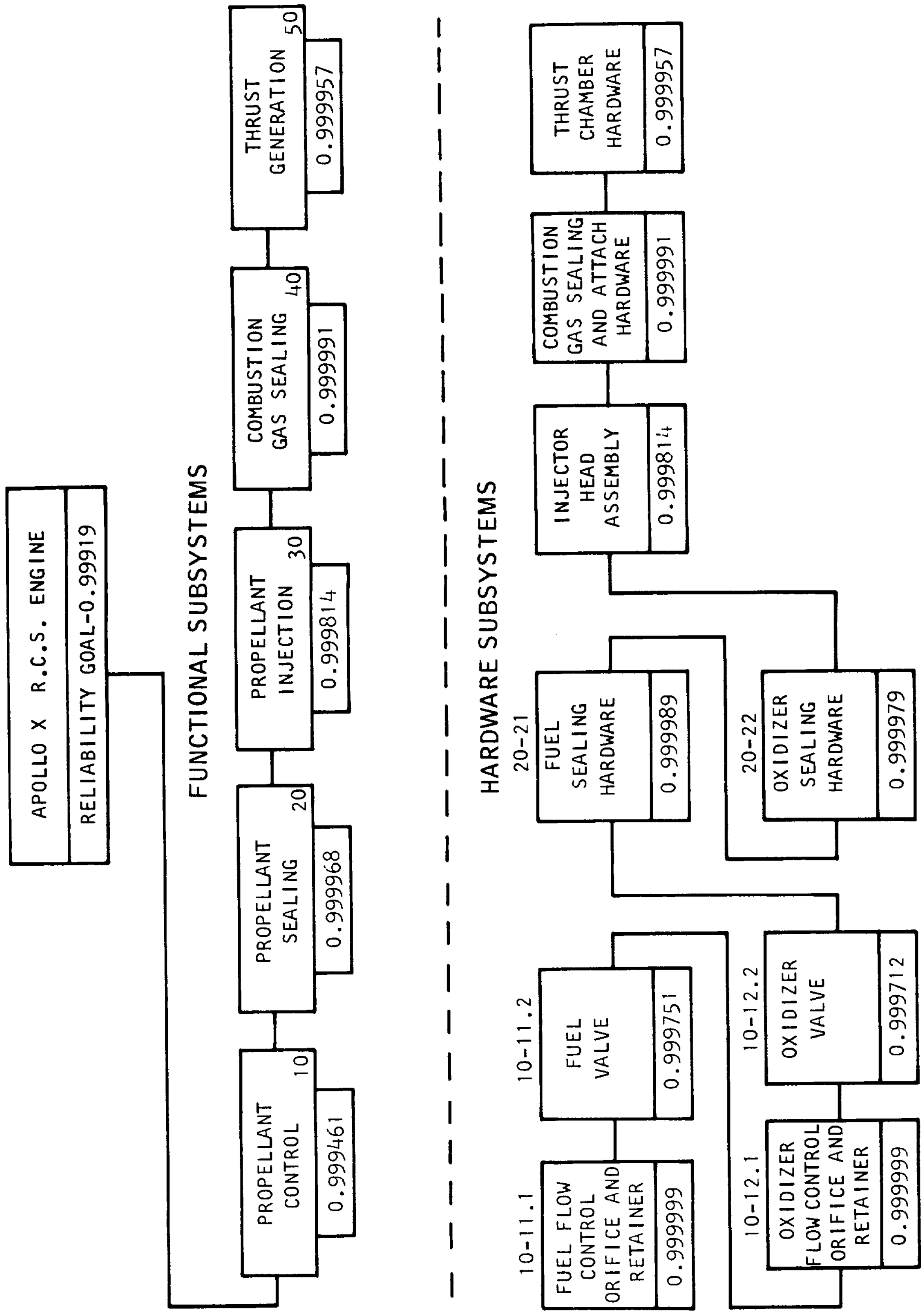


Figure 32

TABLE XI

RELIABILITY FAILURE EFFECT ANALYSIS

APOLLO-X RCS ENGINE

ITEM	FUNCTION	FAILURE MODE	POSSIBLE CAUSES	FAILURE EFFECTS - ENGINE	REMARKS
Solenoid Valves	Permit propellant flow to the combustion chamber when energized with 27 ± 3 volts d.c. and cease flow when de-energized. Response rates and propellant pressure drop must be maintained within the required limits.	1. Cold welding of poppet and seat.	Prolonged exposure to hard vacuum of less than 10^{-9} mmHg.	1. Sluggish valve response with possibility of valve failure to open. 2. Possible loss of engine thrust.	Testing is needed to determine if expected thin propellant film between poppet and seat will prevent this potential problem.
		2. Electrical Short	1. Sublimation of the potting compounds and coil wire insulation during prolonged exposure to hard vacuum of less than 10^{-9} mmHg.	1. Failure of valve to open when energized. 2. Loss of engine thrust.	Testing is needed to evaluate this problem.
			2. Deterioration of cable insulation due to more severe radiation exposure.	1. Failure of valves to open when energized. 2. Loss of engine thrust.	Testing to evaluate the probability of this potential problem is indicated.
		3. Propellant leakage	Teflon seat deterioration from combined effects of more severe radiation exposure, increased propellant exposure, hard vacuum exposure, increased cycles, and long time compressive cold flow. Contamination from the propellants and/or system.	1. Waste of propellant. 2. Possible loss of engine from structural failure due to ignition explosion or from plugged propellant passages resulting from possible icing in the injector or combustion chamber.	Testing is needed to evaluate this problem.
		4. Sticky Poppet	Prolonged exposure to propellants and increased contaminants from increased flow of propellant through the valve may cause the valves to become sticky.	1. Slow opening response or possible failure to open. 2. Possible oxidizer lead if fuel valve is sticky with possible resultant engine loss from ignition explosion.	Lunar Apollo Qual testing may indicate the probability of problems in this area.
		5. Increase in propellant pressure drop	Prolonged propellant exposure may cause sufficient growth of seat material to decrease the stroke and/or change the flow passage discharge coefficient.	Decreased propellant flow rate with corresponding engine thrust degradation.	Testing is needed to evaluate this problem.
Valve - Injector Interface Seals	Prevent external leakage at the interface between the valves and injector head assembly	Teflon seal deterioration.	More severe radiation exposure, increased propellant exposure, hard vacuum exposure, and long time compressive cold flow.	1. External leakage with waste of propellant. 2. Thrust variation and degradation due to decreased propellant flow to the combustion chamber.	Testing needed to evaluate this problem.
Injector Head Assembly	Inject and meter propellants to effect low pressure ignition and to maintain stable and efficient combustion. Injection of fuel from weeper holes gives film cooling to the chamber wall.	1. Plugged injection ports which restrict or prevent proper flow of propellants into the combustion chamber.	Galvanic corrosion during prolonged propellant exposure especially of anodized aluminum in contact with stainless steel.	1. Probable degradation of engine thrust performance. 2. Possible loss of engine from combustion chamber burn-through due to combustion at improper O/F ratio and/or improper propellant distributions. 3. Possible loss of engine due to excessive heating of combustion chamber or attach hardware because of insufficient film cooling.	Redesign to eliminate aluminum - stainless steel interfaces in propellant environments. Thorough testing to evaluate the seriousness of this problem.

TABLE XI (Continued)

RELIABILITY FAILURE EFFECT ANALYSIS

APOLLO-X RCS ENGINE

ITEM	FUNCTION	FAILURE MODE	POSSIBLE CAUSES	FAILURE EFFECTS - ENGINE	REMARKS
Injector Head Assembly (Continued)		2. Structural failure of the valve standoff insert	Prolonged exposure of the Phenolic insulator to the combination of temperature and high vacuum may result in material losses. Since this part is intended to protect the stand-off insert from side loads, over stress may occur with sufficient material loss of the phenolic insulator.	External propellant leakage with possible loss of engine thrust due to excessive propellant leakage.	An evaluation of the effects of prolonged exposure of the Phenolic insulator to temperature and high vacuum is needed with possible testing to support the findings.
Combustion Chamber	Provides a boundary for propellant mixing initial combustion and combustion pressure. Acceleration of the gases through the throat and supersonic expansion in the bell is of primary importance to engine efficiency and thrust generation.	1. Spalling or penetration of the molybdenum disilicide coating.	Probability of spalling of the combustor inside surface or complete wall penetration due meteroid impact is increased because of the extended mission time.	Probable chamber burnout with subsequent loss of engine thrust and/or flame impingement on the spacecraft.	Analysis indicates that a probability of this failure mode not occurring during the 34 day Apollo-X mission is .965 and .9799 for the 45 day mission. Testing is need to document the probability of chamber burnout due to this failure mode. If probability is high, possible design changes required are greater coating thickness, no coating (different base material), or different ductile coating.
		2. Meteroid penetration of the L-605 bell.	Probability of meteroid penetration of the L-605 bell is increased because of the extended mission time.	1. Possible thrust degradation due to bell damage. 2. Possible loss of thrust due to extensive bell damage. 3. Possible flame impingement on the spacecraft.	Analysis indicates that the probability of no meteroid penetration of the L-605 bell is .9524 for the 34 day Apollo X mission and .939 for the 45 day mission. The possibility of this failure mode resulting in any significant reduction in either performance or life characteristics of the engine assembly should be documented by tests.
		3. Wearout of molybdenum disilicide coating.	Increased mission time and increased engine firing time and cycles may result in coating wearout. Repeated cycles of coating sublimation during chamber cooldown time at high vacuum conditions, and oxidation and erosion during hot gas firings may result in a significant reduction of useful chamber life.	1. Oxidation of the molybdenum chamber would result in probable chamber burnout. 2. Probable loss of engine thrust due to loss of combustion chamber pressure.	Documentation of this failure mode must be made to determine the useful life of the molybdenum disilicide coating in a space environment. Design changes to use a better coating or no coating (new chamber material) may be necessary.

IV PROGRAM PLAN

INTRODUCTION

Presented herein is the Apollo X program plan which has as its objective the design, development, qualification, acceptance test, and delivery of reliable hardware suitable for flight usage. The program plan is presented in Figure 33 and contains the following major elements of work:

1. ENGINEERING AND DEVELOPMENT

This portion of the program consists of pre-design, component evaluation, and engine evaluation and development.

a. Pre-Design Phase

This includes specification definition work as well as a program of fundamental laboratory effort to evaluate the materials in their specific environments.

b. Component Evaluation

Evaluation and documentation of reasonable quantities of Apollo components to determine statistically the hardware capabilities and limitations for the Apollo X missions.

c. Engine Evaluation and Development

Evaluation and documentation of the Apollo engine to determine its capabilities and limitations for the Apollo X missions. Also, specific problem areas such as meteoroid penetration, contamination, and thermal management will be fully investigated.

2. QUALIFICATION TESTING

3. RELIABILITY DOCUMENTATION

4. ACCEPTANCE TESTING

5. RELIABILITY SURVEILLANCE AND ASSESSMENT

6. ENGINE DELIVERIES

It is believed that the above elements of work carried out in a comprehensive test program will provide data and documentation evidence that the delivered hardware is indeed reliable and will perform its intended mission as required.

APOLLO X PROGRAM PLAN

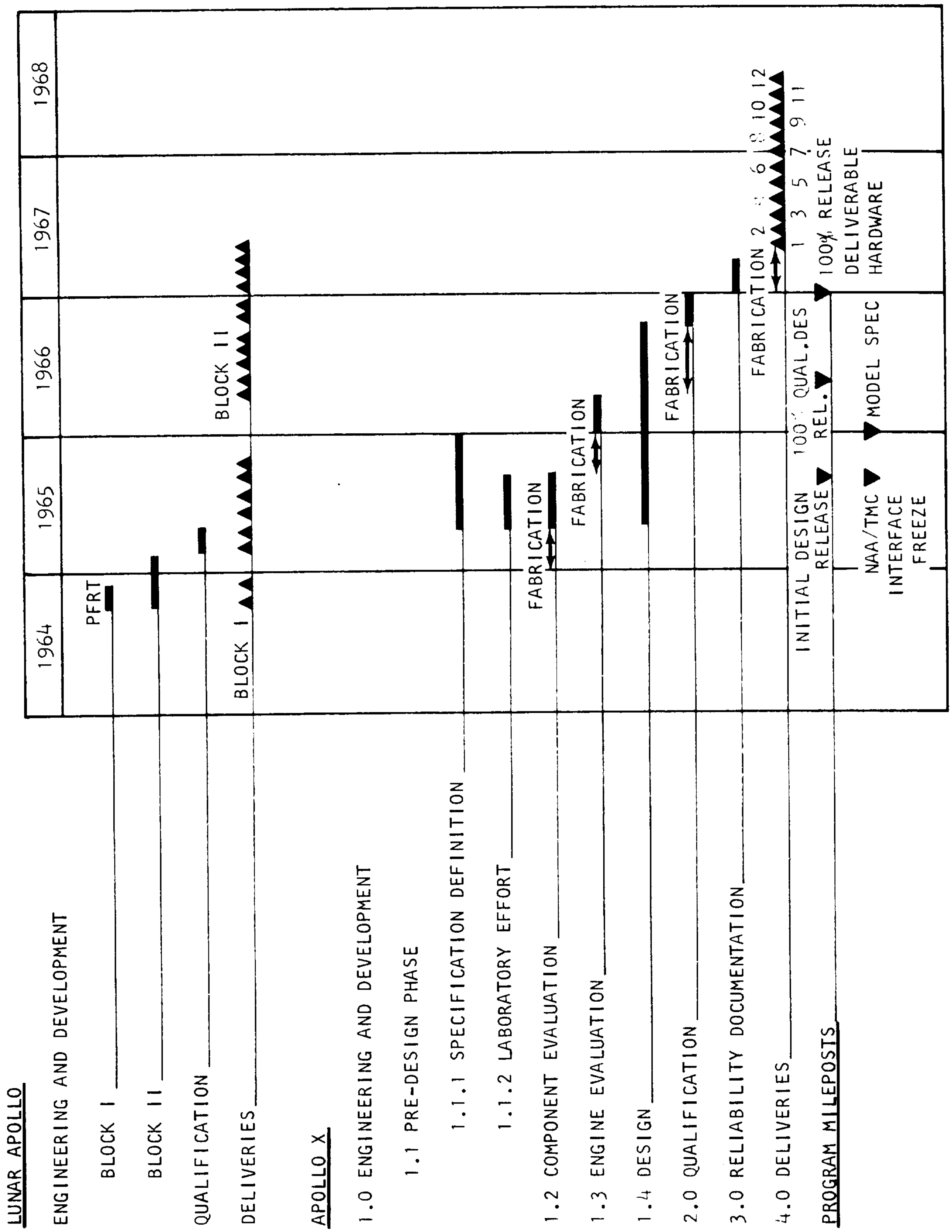


Figure 33

BACKGROUND AND APPROACH

The present Lunar Apollo program provides for the development of a passive thermal control engine through qualification of 5 engines by mid-April 1965. After completion of qualification testing, no additional documentation testing is planned or funded. Since it is the intent of the Apollo X program to use Lunar Apollo hardware (with minimum change) to meet Apollo X requirements, a reasonable amount of additional statistical documentation testing of Lunar Apollo hardware has been included in the Apollo X program plan.

The mission usage of engines on Apollo X will impose the following new requirements on the hardware:

1. Increased exposure time to the space environments and propellants.
2. Increased number of starts.
3. Increased burning time.
4. A new specified duty cycle.
5. Possible differences in the thermal environment surrounding the engine.
6. A requirement for a greatly increased engine reliability goal:

Lunar Apollo Goal	0.997
Apollo X Goal	0.99919

The analytical studies conducted to date indicate that no major problems should be encountered in utilizing the present Apollo engines (with minor redesigns) for the Apollo X mission (with the exception of Item 6 above). However, a reasonably comprehensive test program is required in order to verify the ability of the engines to meet the new requirements.

The test plan presented provides the necessary documentation in the following specific areas of interest:

1. Combustion life as influenced by the effects of:
   a. Hard vacuum (10^{-5+} mm Hg)
   b. Pulsing
   c. Steady state operation.
2. Documentation of preigniter characteristics, minimum impulse repeatability, and reliability over a large number of starts on qualified Lunar Apollo hardware.

3. Effect of engine propellant exposure on subsequent engine performance and operation.

4. Effect of meteoroid impact on combustion chamber life.

5. Effect of contamination on engine performance and operation.

6. Effect on engine life and performance of subjecting engines to specific Apollo X environmental conditions such as high temperature, low temperature, over pressure, under pressure, and then specific Apollo X duty cycle operation.

7. Effect of O/F variations over duty cycle on propellant utilization.

8. Effect of the combined influence of ultra high vacuum and temperature on combustion chamber coating life.

9. Determine the extent of galvanic and simple corrosion of the various metals and/or metal couples planned for use in the Apollo X engine when exposed to propellants for long durations.

10. Effect of a hard vacuum (10^{-9} mm Hg) on valve propellant leakage, cold welding of the seat and poppet, and sublimation of insulation in the valve coil cavity.

11. Effect of long duration hard vacuum on electrical insulation and "O" rings, and seal materials.

12. Effect of prolonged exposure to propellants and prolonged cycling on reasonable quantities of identical valves to determine rates of leakage, valve pressure drop, response, and internal corrosion characteristics.

13. Evaluation of specific Apollo X thermal management problems by engine development tests.

14. Determination of how much margin actually exists in regard to present engine life and ability of present design to meet new reliability goals.

SCHEDULE

The program plan presented in Figure 33 is time phased to provide for delivery of qualified hardware starting in January 1967. In order to achieve this objective the following critical activities must be initiated on the dates noted.

APOLLO X CRITICAL MILESTONES

1.	Fundamental Laboratory Effort Initiated	April 1965
2.	Component Evaluation Effort Initiated	April 1965
3.	Freeze TMC/NAA Design Interfaces	August 1965
4.	Apollo X Full Scale Prototype Engine Test Initiated	January 1966
5.	Final Model Specification to NAA	January 1966
6.	NAA/TMC Review and 100 percent Release of Apollo X Qualified Engine Design	May 1966
7.	Initiate Qualification Test of Apollo X Engine	October 1966
8.	Initiate Reliability Testing of Apollo X Engine	January 1967
9.	NAA/TMC Review and 100 percent Release of Apollo X	January 1967
10.	Delivery of Qualified Apollo X Hardware	June 1967

PROGRAM PLAN OUTLINE

1.0 Engineering and Development

1.1 Pre-Design Phase

1.1.1 Specification Definition

1.1.2 Laboratory Effort

1.1.2.1 Combined Environmental Effects on Chamber Life

1.1.2.2 Compatibility of Materials with Propellants

1.1.2.3 Environmental Effects on Plastics

1.2 Component Evaluation

1.2.1 Injector Valves

1.2.2 Injector Head Assembly

1.3 Engine Evaluation

1.4 Design

2.0 Qualification Test

3.0 Reliability Demonstration

4.0 Fabrication, Acceptance Test, and Delivery of End Item Hardware

5.0 Fabrication, Acceptance Test, and Delivery of Spares

6.0 Reliability

7.0 Quality Control

8.0 Product Support

9.0 Ground Support Equipment

10.0 Special Test Equipment

11.0 Documentation

12.0 Program Management

TASK DESCRIPTIONS

Task 1.1.1

TASK DESCRIPTION

1.0 TITLE

Specification definition phase.

2.0 BACKGROUND

An Apollo X study program is presently in progress which provides for system trade off studies and analyses, and this effort will be included in this task in addition to Specification Preparation and Preliminary Design Effort.

3.0 TASK OBJECTIVE

a) Define Apollo X system requirements and analyze trade-offs between system and engine.

b) Define and prepare an engine model specification.

c) Define design requirements and TMC/NAA interfaces.

4.0 METHOD OF ACCOMPLISHMENT

a) Liaison and coordination with NAA.

b) Analytical studies.

c) Preliminary design studies.

d) Specification preparation and coordination with NAA.

5.0 PROGRAM TIME SPAN

a)	Design requirements	9 months
b)	System trade-off studies	9 months
c)	Specification preparation	9 1/2 months

Task 1.1.2.1

TASK DESCRIPTION

1.0 TITLE

Combined effects of Environment on Chamber Life.

2.0 BACKGROUND

During the transient time period after engine shutdown, the combustion chamber pressure decreases rapidly relative to the wall temperature decrease. During the time at which the walls are relatively hot, a significant amount of silicon could be lost from the disilicide coated combustor. Subsequent exposure to an oxidizing atmosphere further degrades the coating. Repeated cycles of vacuum exposure and exposure to combustion gases could result in a useable life less than that which would result from coating oxidation on the inside surface during continuous engine operation.

3.0 TEST OBJECTIVE

Support engine combustion chamber life tests by conducting a laboratory evaluation to determine the effects on combustor life of exposing samples to a combination of hard vacuum, oxidizing atmosphere, and space radiation. Determine the type of cycling between hard vacuum and oxidizing atmosphere which is the most detrimental to coating and chamber life so as to provide data to be used in full scale engine tests.

4.0 METHOD OF ACCOMPLISHMENT

a) Maintain samples at a representative transient cool down temperature (i.e., 2000°F) for several minutes at 10^{-5} mm Hg, corresponding to several cool down cycles.

b) Subsequently heat samples to 2350°F in an oxidizing atmosphere.

c) Repeat a and b above several times and determine rate of coating and material degradation.

5.0 HARDWARE AND EQUIPMENT REQUIRED

a) Vacuum chamber to product 10^{-5} mm Hg.

b) Fifty-four (54) molybdenum tubes.

c) Thirty-six (36) molybdenum discs.

d) Coating services.

e) Oxy propane.

The torch tests and emittance tests will be performed in Building L-14.

6.0 PROGRAM DURATION

Five months.

Task 1.1.2.2

TASK DESCRIPTION

1.0 TITLE

Compatibility of Materials with Propellants

2.0 BACKGROUND

Evidence acquired at Marquardt shows that galvanic corrosion can occur if dissimilar metal couples are exposed to wet N_2O_4 or wet hydrazines. Since under field conditions some air and some moisture may become entrapped when the propellant systems are filled, galvanic corrosion could occur during a prolonged space mission.

3.0 TEST OBJECTIVES

The objectives of this program are:

a) To measure the extent of galvanic or simple corrosion of various metals and/or metal couples exposed to propellants.

4.0 METHOD OF ACCOMPLISHMENT

Metal specimens and metal couple specimens will be prepared and immersed in propellant. The propellants will be removed and replaced with fresh propellants every twenty days of the test program. The propellants will be analyzed for propellant decomposition products and for corrosion products. The corrosion specimens will also be tested for weight change and intergrannular attack at welds, etc.

5.0 WHAT IS NEEDED TO ACCOMPLISH PROGRAM

a) Ten glass lined exposure chambers.

b) Propellants chambers.

c) Two pressure vessels (1 quart autoclaves) for exposure of valves.

d) Analysis of propellants and specimens will be done in M&P Laboratory in Building L-14.

Task 1.1.2.2

The following metals and materials will be utilized to prepare galvanic couples or for simple corrosion testing:

Aluminum Alloy	6061, T-6 with anodized costs. 6061, T-6 with Martin "Hard Coat" or equivalent.
Steel Alloy	304 stainless steel 446
Steel Alloy	446 Electrolyzed A-286 PH 15-7 Molybdenum
Nickel Alloy	Inconel X Inconel 92 welds
Coated Alloys	17-4 PH with FEP coating 15-7 PH molybdenum with TFE coating
Non Metallic Material	glass filled teflon

6.0 PROGRAM DURATION

Five months.

Task 1.1.2.3

TASK DESCRIPTION

1.0 TITLE

Combined Effects of Temperature and High Vacuum Environments on Plastics.

2.0 BACKGROUND

Present literature indicates that the combined environments of high vacuum and temperature can produce significant mechanical and physical property changes in plastics and elastomers.

3.0 TEST OBJECTIVE

To determine the effects of combined high temperature on the following materials and components:

a) Electrical insulating tapes
b) Potting compounds
c) Cable insulations
d) Solenoid valves
e) "O" rings
f) Seals
g) Gaskets
h) Corrosion pairs of insulations and seals with structural materials

4.0 METHOD OF ACCOMPLISHMENT

The materials, combinations, and components samples will be tested in an ultra high vacuum and temperature chamber. After exposure, the materials will be subjected to electrical, microscopic, and/or mechanical tests to determine what effects, if any, were caused by the exposure to vacuum and temperature.

5.0 WHAT IS NEEDED TO ACCOMPLISH TASK

Equipment - Ultra High Vacuum Chamber with temperature control (includes pumps and gages, etc.). Magohmmeter - available. Standard Lab Equipment - available. All electrical, microscopic, and mechanical tests will be performed in Building L-14.

6.0 PROGRAM DURATION

Five months.

Task 1.2.1

TASK DESCRIPTION

1.0 TITLE

Injector Solenoid Valve

2.0 BACKGROUND

Performance characteristics and compatibility of the Apollo solenoid valves (P/N X21427 fuel and X21428 oxidizer) for 34 and 45 day missions have been reviewed. In general, it may be stated that the valves are suitable for 45 day missions except that actual test data on the valves at better than 10^{-9} mm Hg vacuum does not exist and data on propellant exposure and valve cycling while exposed to propellants is insufficient. Consequently, it is recommended that additional testing of the valves be conducted to verify operation under hard vacuum and during prolonged propellant exposure.

3.0 TEST OBJECTIVES

a) Determine the effects of hard vacuum (10^{-9} mm Hg) on propellant leakage, cold welding of the seat and poppet, and sublimation of insulation in the coil cavity.

b) Determine the effect of prolonged propellant exposure on leakage, valve pressure drop, valve response, and internal corrosion.

4.0 METHOD OF ACCOMPLISHMENT

Recommended Test Program

The proposed test program is to evaluate the effects of three types of environments on valve performance. These are radiation, hard vacuum, and propellant exposure.

The hard vacuum and propellant environments may affect several performance aspects of the valve. Consequently, a test maxtrix is proposed which will utilize several valves during the test program. Performance characteristics will be verified as follows:

a) Leakage

Fill valve with propellant and pressurize to 181 psi. Raise valve temperature to 150°F. Cap valve upstream and monitor pressure in the valve while the valve outlet is subjected to a vacuum of 10^{-9} mm Hg.

b) Cold Welding

Take a dry, clean valve and subject it to 10^{-9} mm Hg vacuum while heating it to 250°F. Actuate valve several times and monitor response. Allow valve to remain in vacuum at 250°F for seven days.

Actuate and verify response. Allow valve to remain in vacuum and heat for fourteen days. Again verify response. Repeat entire procedure with 181 psi GN_2 in valve.

c) Sublimation

After an exposure to vacuum of more than 45 days, subject valves to electrical checks per MTP 0002. Compare data with pre-exposure data.

d) Sticking

Allow valve to remain in a vacuum for more than 45 days while filled with propellant and pressurized to 181 psi. Energize valve and monitor response.

e) Pressure Drop

Subject valve to rated propellant flow (or water flow) and measure ΔP.

f) Endurance

Cycle valve the predetermined number of times while monitoring response and with propellant in the system.

g) Corrosion

Disassemble valve and inspect sliding and seating surfaces for possible wear.

h) Drain and Purge

Drain valves of propellant and purge with GN_2. Allow valve to remain for fourteen days. Fill with propellant again, actuate several times, purge with GN_2 and allow valve to remain for one month. The following test matrix is proposed. (Letters after valve number signify oxidizer or fuel. Numbers in the matrix indicate order of test in the numerator and the denominator indicates if the particular test is to be performed more than once.)

Task 1.2.1

TEST

VALVE NUMBER

	10	20	30	40	5F	6F	7F	8F	90
A	2 4 6 9 11	2 4 6 9 11	1 5 10	1 5 10	2 4 6 9 11	2 4 6 9 11	1 5 10	1 5 12	1 5 12
B	1	1			1	1			
C			7	7			7	7	7
D			2	2			2	2	2
E	7	7	3 6	3 6	7	7	3 6	3 6	3 6
F	3/1 5/10 8/1K 10/100K	3/3 5/50 8/5K 10/1M	4/5 9/1M	4/50 9/1M	3/1 5/10 8/1K 10/100K	3/3 5/50 8/5K 10/1M	4/5 9/1M	4/50 11/1M	4/50 11/1M
G	12	12	8 11	8 11	12	12	8 11	8 10 13	8 10 13
H								9	9

5.0 WHAT IS NEEDED TO ACCOMPLISH TASK

a) Hardware required - 9 injector valve assemblies, plus spare parts.

b) Ultra high vacuum chamber 10^{-9} (not available at TMC - to be done outside).

c) Set up for soak and cycling test (may also be done outside so it is compatible with vacuum test).

6.0 PROGRAM DURATION

Four months.

Task 1.2.2

TASK DESCRIPTION

1.0 TITLE

Injector Head Assemblies Evaluation (Soak and Cycle)

2.0 BACKGROUND

The Apollo X requires that the injector head which has a combination of steel and aluminum parts will see additional propellant exposure time up to 45 days. The test proposed here will evaluate the effect of the additional exposure on injector and engine performance and corrosion generation.

3.0 TEST OBJECTIVE

a) Evaluate the effect of the increased propellant exposure time on injector performance and corrosion generation.

4.0 METHOD OF ACCOMPLISHMENT

a) Calibrate three injector heads with H_2O.

b) Conduct a 45 day soak and cycle test of injector head assembly soaked in N_2O_4 and one in Aerozine 50. Cycle valves one cycle every other day for 15 days then no cycling just exposure for 30 days.

c) Recalibrate injector heads.

d) Disassemble two injectors and inspect for corrosion.

e) Leave third head assembled, clean purge and store for 30 days and then conduct engine performance evaluation as specified in Task 2.4.

5.0 HARDWARE REQUIRED

Three injector head assemblies set up.

6.0 PROGRAM TIME DURATION

Three months.

Task 1.3

TASK DESCRIPTION

1.0 TITLE

Apollo X Engine Development Tests.

2.0 BACKGROUND

The Apollo X mission imposes the following new requirements on the engine combustor:

a) Increased burning time.

b) Increased number of starts.

c) Increased exposure to a hard vacuum while at operating and transient cool down temperatures.

d) An increased engine reliability goal.

These conditions impose the following possible failure modes on the combustor:

a) Failure due to time versus temperature oxidation of the chamber and coating.

b) Failure due to multiple cycles of high strain rate starts.

c) Failure due to sublimation of the coating in a hard vacuum and subsequent exposure to an oxidizing atmosphere.

d) Failure due to a combination of a, b, and c above.

In addition to the need for data on the combustor, the following additional engine information is needed in order to determine capability of the engine to do the Apollo X job.

a) Documentation of preigniter characteristics and minimum impulse bit repeatability and reliability over a large number of starts on qualified Lunar Apollo hardware.

b) Determination of how much margin actually exists in terms of overall engine operating life and ability of present design to meet new reliability goals. This includes valve operation, heat soak back, thermal cycling, and all the other effects which are evaluated when operating a full scale engine.

c) Determination of the effect of meteoroid impact on combustion chamber life.

d) Effect of contamination on engine performance and operation.

e) Evaluation of specific Apollo X thermal management problems by engine development tests.

3.0 TEST OBJECTIVES

a) Determine the effects of 45 day propellant exposure on subsequent engine performance and operation.

b) Determine the effect of contamination on subsequent engine performance and operation.

c) Determine the effects of meteoroid penetration on engine operating life.

d) Conduct specific Apollo X duty cycle tests combined with other environmental and performance effects such as:

1) Over pressure operation
2) Under pressure operation
3) High temperature operation
4) Low temperature operation
5) Off O/F operation

e) Conduct specific thermal management tests which are required due to changes in the thermal environment of Apollo X versus Lunar Apollo.

4.0 METHOD OF ACCOMPLISHMENT

a) 45 Day Propellant Exposure and Shelf Life

Take one of the injector heads which has been used in the injector head soak and cycle tests, Task 1.2.1, and after it has been cleaned and sets on the shelf for an additional 30 days, assemble it into an engine and conduct a performance evaluation test to see what effect the soak and cycle and shelf life have had on the engine's performance and operation.

b) Contamination Test

Phase A

1) Water flow injector head.

2) Attach combustion chamber and install in a vertical up position.

3) Conduct a simulated launch in a blow down facility. Flow unfiltered air straight into the nozzle exit.

4) Remove chamber and inspect.

5) Reflow injector head and disassemble and inspect.

Phase B

1) Repeat 1 and 2 of Phase A.

2) Conduct sand and dust test on engine without any protection covers installed. Sand-dust per Mil Standard 810.

3) Reflow check, inspect, and disassemble.

Phase C

Based on the results of Phases A and B, a burn test may be required to determine the effects of A and B on engine operation and performance.

Phase D

Conduct a burn test on an engine with one of the main chamber bleed holes plugged and one of the preigniter holes plugged and evaluate effect on engine operation.

c) <u>Meteoroid Test</u>

Make small holes in the combustion chamber at various locations and run engine tests to evaluate the effect on operating life.

5.0 <u>HARDWARE</u>

Five engines and spares.

6.0 <u>PROGRAM TIME DURATION</u>

Three months.

Task 2.0

TASK DESCRIPTION

1.0 TITLE

Qualification Tests

2.0 BACKGROUND

There is sufficient difference between the Lunar Apollo requirements and Apollo X requirements in regard to burning time, multiple starts, propellant and space exposure time, specific duty cycle to require a formal qualification demonstration test on the Apollo X engine design.

3.0 TEST OBJECTIVE

a) Demonstrate through a formal test the capability of the Apollo X engine to meet its specification requirements.

4.0 METHOD OF ACCOMPLISHMENT

Conduct testing on five engine assemblies which will be essentially identical to that presently called out in Paragraphs 4.6, 4.7, and 4.8 of the present NAA specification MC 901-0004-D revision, dated 17 August 1964 except the specific duty cycles, etc., will be altered to reflect Apollo X requirements.

The testing to be conducted will include:

1) Calibration test
2) Shock (transportation)
3) Vibration (transportation)
4) Humidity
5) Salt fog
6) Static load (limit)
7) Vibration (boost)
8) Electrical and structural integrity
9) Vibration (space flight loading)
10) High temperature - vacuum
11) Mission simulation
12) Pulse operation survey
13) Temperature (cycling)
14) Mission simulation
15) Orbit retrograde (continuous run)
16) Pulse operation survey
17) Direct coil duty cycle
18) Calibration

19) Electrical and structural integrity
20) Corrosion resistant (oxid-fuel)
21) Final examination
22) Off limits test (3 engines)
23) Life tests (2 engines)

As shown on the program plan, Figure 1, a NAA/TMC review will be held in December 1965, prior to 100 percent qualified design, to review the results of the documentation program and the changes in design which are to be incorporated into the qualified engine.

At this point in time, those qualified tests already completed on Lunar Apollo which are not affected by the design changes presented may be eliminated from the qualified plan and need not be repeated.

5.0 HARDWARE REQUIRED

Five engines plus one spare.

6.0 PROGRAM DURATION

Three months.

Task 3.0

TASK DESCRIPTION

1.0 TITLE

Reliability Documentation

2.0 BACKGROUND

A much more severe reliability requirement (0.99919) is imposed for the 34 day Lunar Polar Orbit mapping mission for Apollo X than for the Lunar Apollo mission (0.997). The reliability requirement for Lunar Apollo is only a goal and no program is presently funded to demonstrate this reliability goal.

3.0 TEST OBJECTIVE

To demonstrate a reliability of 0.99919 at a confidence level of 60% for the Apollo X engine for the critical failure modes due to the combined effects of hard vacuum (10^{-5} mm Hg), pulsing, and steady-state operation.

4.0 METHOD OF ACCOMPLISHMENT

Five complete qualified engine assemblies will be subjected to the following sequence of tests:

a) Soaking of the combustor in a hard vacuum 10^{-5} mm Hg for a specified time while the combustor is heated to approximately 2000°F.

b) Pulsing of the complete engine for a specified period of time at altitude conditions 100,000 feet plus.

c) Steady state operation of the engine for a specified time.

d) Repeat the cycle a, b, and c above several times as determined by results of lab investigation.

The above tests shall be continued until the specified times are accumulated or failure, whichever occurs first.

For Each Engine

Total burning time accumulated	9,760 seconds
Total starts accumulated	142,125
Total vacuum soak at temperature	10,860 seconds

The total times accumulated above on each engine are equivalent to over 12 Apollo X duty cycles and if all engines complete the test without failure, will document the hardwares' capability to meet the life goal of a reliability of 0.99919 with a 60 percent confidence as far as the variables investigation are concerned.

5.0 HARDWARE REQUIRED

Five complete engine assemblies
Two spare combustion chambers
Additional spare parts

6.0 PROGRAM DURATION

Three months.

Task 4.0

TASK DESCRIPTION

1.0 TITLE

Fabrication, Acceptance Test, and Delivery of End Item Hardware

2.0 BACKGROUND

The present Apollo X program shows that 12 ship sets (16 engines each) of end items hardware will be required to be delivered to the customer for use in house and test spacecraft as well as flight vehicles.

3.0 OBJECTIVE

To fabricate, acceptance test, and delivery quality end item hardware conforming to specification requirements to the customer on or before the negotiated delivery dates.

4.0 METHOD OF ACCOMPLISHMENT

4.1 Release the design that has successfully completed the Qualification Test.

4.2 Fabricate, assemble, inspect, acceptance test, package, and ship the number of engines as finally negotiated.

4.3 Provide those recurrent services as required such as engineering support of manufacturing and program management required to supervise the above activities.

5.0 PROGRAM DURATION

Twenty-five months.

Task 5.0

TASK DESCRIPTION

1.0 TITLE

Fabrication, Acceptance Test, and Delivery of Spares

2.0 BACKGROUND

Delivery of spares will be required to support the customer requirements that will be outlined in the Apollo X logistics and support effort.

3.0 OBJECTIVE

To fabricate, acceptance test, and deliver quality hardware conforming to specification requirements to the customer on or before the negotiated delivery dates.

4.0 METHOD OF ACCOMPLISHMENT

4.1 Release the design that has successfully completed the Qualification Test.

4.2 Fabricate, assemble, inspect, acceptance test, package, and ship the required hardware.

4.3 Provide those recurrent services as required such as engineering support of manufacturing and program management required to supervise the above activities.

5.0 PROGRAM DURATION

Twenty-five months.

Task 6.0

TASK DESCRIPTION

1.0 TITLE

Reliability Program

2.0 BACKGROUND

A reliability program has been implemented with performs and/or assures the performance of the reliability disciplines, tasks, and management efforts required to attain, estimate compliance with, and document the attainment of a highly reliable Apollo Reaction Control Engine. These activities are in general conformance to the program requirements established by North American Aviation and MIL-R-27542, NASA 200-2, and applicable second-tier specification.

3.0 TASK OBJECTIVES

The reliability program serves as the basis for:

a) assure that the engine and interface designs have the inherent performance capabilities required to meet reliability requirements;

b) demonstrate by a statistically designed test program that the system meets the reliability requirements;

c) monitor the test program and test results;

d) identify and isolate causes and eliminate and prevent the recurrence of failures, malfunctions, or discrepancies capable of degrading performance; and

e) assure that the methods used in manufacturing, processing, and logistic handling of the system will maintain inherent design qualities without degradation.

4.0 METHOD OF ACCOMPLISHMENT

The Reliability Program will encompass the activity areas of the Reliability Engineering, Reliability Assurance, Hi-Reliability Standards, and Reliability Management.

Task 6.0

a) Reliability Engineering

1) Design Reviews

Detailed reliability design reviews will be performed; problem areas and potential modes-of-failure will be called out, and special test requirements to ensure reliability will be determined. Corrective action requests will be initiated and followed up, where required. The design reviews will include consideration of human engineering, maintainability, safety, and interchangeability factors, as well as functional and strength adequacy, material compatability, environmental effects, fabrication methods and inspection adequacy, test program adequacy, configuration control, packaging, handling, storage, and logistic effects.

2) Reliability Analyses

The engine system numerical reliability goals will be apportioned to the subsystem; a mathematical model will be established quantitatively describing the components of failure; reliability optimization and safety margins studies will be performed; progress in attainment of performance goals will be monitored.

3) Integrated Performance and Reliability Test Planning Analysis

The test plans for each system phase will be reviewed. Reliability data requirements will be integrated into the test specifications; tests will be monitored; and test data will be analyzed for variance from design objectives.

4) Malfunction/Failure Investigations

Malfunction and failures will be reported and malfunction/failure analyses and corrective action will be instituted on all failures in order to identify, isolate, and eliminate the causes.

5) Design and Specification Tolerance and Safety Margins

Data and documentation generated by Engineering Analysis, Design, Manufacturing, Test, Inspection, etc., will be reviewed and assessed to establish design, processing, and test performance tolerances, safety margins, specification limits, and acceptance limits on critical characteristics parameters.

Reduced test data will be analyzed to assess variance from design; performance will be correlated with reliability prediction indices; reliability estimates will be prepared, periodically upgraded, and reported.

b) <u>Reliability Assurance</u>

Pertinent data from tests and test inspections will be tabulated, analyzed, and correlated. Data will be analyzed to extract a variety or relationships desired for reliability prediction and demonstration. Dispersion (spread) studies will be performed to determine conformity with the assumed probability distribution to assure validity of the appropriate reliability analyses.

c) <u>Hi-Reliability Standards</u>

1) Specifications and Standards "Call-Outs" on drawings will be reviewed to assure: (1) adequate call-out of requirements and tolerances for materials, components, processing, and procedures; (2) that the design is reproducible; and (3) that components are satisfactory for intended use.

2) Component qualification requirements will be developed sufficiently to meet hi-reliability standards and the allocated reliability requirements of the subsystem.

3) Suppliers/vendors will be controlled and monitored to assure the obtaining of qualified parts.

4) Component qualified status will be monitored, assessed, and reported based on the results of appropriate test programs.

d) <u>Reliability Management</u>

1) Reliability program schedule will be established, correlated with Project activities schedule. Milestone charts will be prepared, upgraded as required, and progress reported.

2) All phases of the reliability program will be monitored and assessed, and reliability status reported. Monitoring techniques will include program progress in developing specifications, standards, and tolerance limits; test reliability demonstration, and malfunction and failure status.

3) A project data control center will be established for the tabulation, storing, summarizing, and retrieving of pertinent data.

4) Corrective action logs will be established, corrective action boards convened, and closed-loop follow-up performed on all deficiencies affecting critical components, characteristics, or performance parameters.

5) Reliability reports will be submitted on varying schedules covering activities and program progress, updated program plans, reliability estimates and assessments, results of failure investigations and analysis, reliability problem areas, and the results of special reliability studies.

5.0 PROGRAM DURATION

Thirty months.

Task 7.0

TASK DESCRIPTION

1.0 TITLE

Quality Control Program

2.0 BACKGROUND

A Quality Control Program has been implemented which complies with the intent of both NPC 200-2 and MIL-Q-9858 as well as applicable second-tier documents for the Apollo Reaction Control Engine. The Quality Control System has been surveyed many times and has been approved by the cognizant government inspection agency which in this case is the USAF.

3.0 TASK OBJECTIVES

To define, identify and isolate cause of, eliminate and prevent the recurrence of configurational and processing discrepancies capable of causing failure or degrading performance.

4.0 METHOD OF ACCOMPLISHMENT

A Quality Control Program Plan is implemented which is designed to assure that:

a) specific descriptive procedures and instructions are prepared;

b) sub-contracted supplies are adequately controlled;

c) measuring and test equipment are periodically calibrated;

d) receiving and in-process inspection operations are performed to detailed inspection instructions;

e) stringent controls are maintained over discrepant materials to:

   1) prevent their use; and
   2) minimize the probability that deficiencies will occur;

f) review board activities are maintained and controlled;

g) adequately detailed documentation is recorded and tabulated;

h) the data accumulated is analyzed and fed back to the appropriate area of activity; and

i) corrective action taken on deficient materials and processes are instituted promptly and efficiently.

5.0 PROGRAM DURATION

Thirty months.

Task 8.0

TASK DESCRIPTION

1.0 TITLE

Product Support

2.0 OBJECTIVE

To provide assistance and/or support required for all launch, field, and off-site operations concerned with the rocket engine.

3.0 METHOD OF ACCOMPLISHMENT

3.1 Assist the customer in off-site system ground tests, by providing assistance in preparation of test plans, the conduct of these tests, and the analysis of the test results with respect to the rocket engines performance.

3.2 To recommend test operations and test procedures required during field and launch checkout.

3.3 Assist in pre-flight preparation, pre-flight check-out, flight and post-flight analysis and flight data.

3.4 Coordinate overhaul, repair, modification and/or retro-fit of customer returned hardware.

3.5 Define and implement the required logistics planning (spares, parts-break manual, training manual and courses, technical handbooks) as negotiated with the customer.

4.0 PROGRAM DURATION

Forty-two months.

Task 9.0

TASK DESCRIPTIONS

1.0 TITLE

Ground Support Equipment

2.0 OBJECTIVE

To analyze and define the ground support equipment requirements for Apollo X. Subsequent effort would be required to design, fabricate, and check-out the negotiated ground support equipment requirements.

3.0 METHOD OF ACCOMPLISHMENT

3.1 Determine which portions of the system will be subject to performance degradation, levels of replacement, interchangeability, criticality, MTBF, etc.

3.2 Draw conclusions regarding service required, tests required, parameters to be tested, equipment accuracy, and items of GSE.

3.3 Using information on process time limitations, personnel skill, levels, environment, and facility and system interface details; establish equipment degree of automation, type of readout, packaging, etc.

3.4 Prepare Ground Support Equipment specifications, maintaining close liaison and interchange of planning information with NAA/S&ID.

3.5 Design, fabricate, checkout, and deliver the required Ground Support Equipment as negotiated with NAA/S&ID.

4.0 PROGRAM DURATION

Thirty months.

Task 10.0

TASK DESCRIPTION

1.0 TITLE

Special Test Equipment

2.0 OBJECTIVE

To define, design, fabricate, and checkout the necessary special test equipment required to conduct all phases of testing.

3.0 METHOD OF ACCOMPLISHMENT

3.1 Determine the special test equipment required to conduct the tests per the detailed test plan requirements.

3.2 Design, fabricate, and install the required special test equipment.

3.3 Conduct the necessary checkouts required to determine the acceptability of the special test equipment for the test phase requirements.

4.0 PROGRAM DURATION

Eighteen Months

Task 11.0

TASK DESCRIPTION

1.0 TITLE

Documentation

2.0 BACKGROUND

Documentation will consist of those requirements to be specified by NAA/S&ID. It is assumed that these documentation requirements will be similar to those outlined in North American Document MC 999-0025, "Documentation Requirements for Apollo Major Subcontractors, General Specification for"

3.0 OBJECTIVE

To provide for all of the documentation and reporting requirements as specified in the negotiated work statement.

4.0 METHOD OF ACCOMPLISHMENT

The following is a list of some of the documents that are assumed to be required for delivery to NAA/S&ID per the requirements and dates to be negotiated:

Program Plan, PERT, Progress Reports, Final Report, Resume of Failure Reports, Final Reliability Report, Final Model Specification, Drawings, Financial Reporting, Quarterly Report, Reliability Program Plan, Acceptance Test Plan and Specifications, etc.

5.0 PROGRAM DURATION

Forty-two months

Task 12.0

TASK DESCRIPTION

1.0 TITLE

Program Management

2.0 OBJECTIVE

To implement and maintain effective program management through all phases of effort as required by the NAA/S&ID statement of work that will be negotiated.

3.0 METHOD OF ACCOMPLISHMENT

3.1 Direct the development of detailed program plans, schedules, and budgets.

3.2 Direct the preparation of clear and concise planning and control documents, work orders, and other instructions consistent with program objectives and limitations.

3.3 Monitor engineering effort and other line organizations to assure compliance with program plans and objectives.

3.4 Initiate corrective measures to prevent or eliminate variances between actual and planned performance.

3.5 Assist in making technical decisions as required.

3.6 Maintain communications with NAA/S&ID and integrate their requirements and Marquardt divisional efforts.

3.7 Maintain a continuing awareness of existing and future NAA/S&ID requirements.

4.0 PROGRAM DURATION

Forty-two months.

APPENDIX A

COMBUSTOR LIFE AT VACUUM CONDITIONS

SUMMARY

Failure modes for the disilicide coated molybdenum combustor are investigated. It is predicted that the coating weight loss due to sublimation in hard vacuum for 600 days while the engine is not operating is negligible. The small amount of coating sublimation which should occur on the outside surface of the combustor during engine operation should not significantly affect engine life. It is postulated that combustor failure can occur because of a coating wearout failure during continuous engine operation and also because of alternate cycles of interior coating sublimation during engine cool down, which removes the oxidation resistant SiO_2 film, and the subsequent high oxidation environment of the combustion gases. The former failure mode can be predicted from engine tests at altitude. Combustor life should exceed one day of continuous engine operation based on current test data. Documentation of the latter failure mode requires laboratory and/or combined laboratory-engine testing to provide alternate cycles of high vacuum and combustion gas simulation. A recommended test program is outlined.

INTRODUCTION

The work conducted to date on determining the characteristics of silicide coatings on molybdenum has been for the purpose of obtaining answers to specific questions, such as those concerning weight loss and emittance in a vacuum at high temperature versus time, oxidation and attack by gaseous mediums, and physical and mechanical properties. Sufficient laboratory work has not been conducted to document all of the disilicide coating characteristics pertinent to a multi-start radiation cooled engine operating in a space environment. Marquardt has determined the emittance versus time of machined and grit blasted molybdenum tubes with different disilicide coatings at temperatures from 2900°F to 3200°F in vacuums of 1-10 microns of Hg (10^{-3} - 10^{-2} mm Hg) and 0.05 psia (2580 microns) of air pressure. (References 1, 2, and 3). These studies were conducted to aid in determining the effect on wall temperatures of evaporation and/or chemical change of the coating on the outside of the thrust chamber. Hughes Aircraft Company conducted tests for North American Aviation to determine whether Durak B coating on molybdenum vaporizes to the extent of endangering the functioning of the thrust chamber and whether vaporization products would deposit on near-by thermal control surfaces. (Reference 4). The tests were conducted at temperatures from 2900°F to 3300°F at pressures of about 10^{-7} torr (mm Hg) for times up to 40 minutes. Lockheed Missiles and Space Company (LMSC) is conducting laboratory studies on coatings for refractory metals in aerospace environments for a broad range of anticipated performance requirements. The tests to date have been conducted to determine sample weight loss rate versus time at temperatures of 2600°F to 3200°F at vacuums of about 10^{-5} mm Hg air pressure for times up to four hours. (References 5 and 6). Climax Molybdenum Company has gathered and summarized physical, mechanical, and chemical properties of three stable molybdenum silicides ($MoSi_2$ with 36.9% Si, Mo_5Si_3 with 14.9% Si, and Mo_3Si with 8.9% Si). (Reference 7).

The chemical and physical changes which occur to a disilicide coated molybdenum thrust chamber at elevated temperatures are different from those caused by exposure to a hard vacuum and to an oxidizing atmosphere. At one atmosphere of air pressure at 2700°F, LMSC reports that only significant structural change from the original $MoSi_2$ coatings in four hours is the formation of a thick zone of Mo_5Si_3 in the middle and a thin layer of Mo_3Si at the metal interface by diffusion. Only a thin zone of $MoSi_2$ remains at the surface. Coating thickness increases. After four hours at 2700°F in air at 0.1 mm Hg, a spongy mixture of Mo (molybdenum), Mo_3Si, and SiO_2 (glass) formed on the surface. All of the $MoSi_2$ had been consumed by diffusion and oxidation, with a large decrease in coating thickness. The coating changed from $MoSi_2$ to Mo_5Si_3 with a thick layer of Mo_3Si at the coating metal interface (Reference 6). At low pressure, the coating degradation is greater at 2600°F than at 2700°F. In four hours at 0.1 mm Hg, all of the coating is converted to a spongy layer at 2600°F while only about half of the coating has been converted to sponge at 2700°F. At temperatures above 2600°F, Mo_5Si_3 is capable of developing its own protective film of SiO_2, while at 2600°F and lower, a protective glass does not form and the coating is rapidly oxidized. Therefore, when sufficient loss of silicon, Si, occurs to convert the coating to Mo_5Si_3, rapid oxidation occurs at 2600°F and lower. The spongy Mo - glass layers formed at low pressures protect the substrate but results in failure instantly on increasing the pressure at the same temperature. Essentially the same explanation is presented in Reference 7: the good oxidation resistance is due to a vitreous SiO_2 film which forms on the $MoSi_2$. This protective film is the result of volatilization of the oxidation product of the molybdenum in the form of MoO_3, leaving behind the oxidation product of the silicon, SiO_2, to form an adherent film of high oxidation resistance. Accelerated oxidation at 900°F to 1650°F occurs because the MoO_3 cannot volatilize in this temperature range.

The resistance of $MoSi_2$ to oxidizing atmospheres is similar to that in air although the weight change may be higher. The atmospheres include CO_2, water vapor, and NO_2. The resistance is less in nitrogen, argon, protective gases, and CO. Hydrogen and hydrogen containing gases can be harmful is no oxygen is present.

The coating weight loss (and degradation) in a vacuum at temperature is most likely caused by "disproportionation" of the silicides. The formation of the Mo_5Si_3 cap (which is less oxidation resistant than $MoSi_2$) is caused by the reaction:

$$5\ Mo\ Si_2 = Mo_5\ Si_3 + 7\ Si\ (gas)$$

At 3000°F, this compound in turn disproportionates to Mo_3Si and Si (gas), forming a thick cap of Mo_3Si above the Mo_5Si_3 layer. At 3200°F, the Mo_3Si layer disproportionates to Mo and Si (gas) forming a Mo cap at the surface. LMSC reports that exposure to vacuum for as little as 30 minutes at 2600°F will cause significant coating degradation on subsequent exposure to air.

DISCUSSION

A disilicide coated combustor which has passed the acceptance tests could possibly fail in one or a combination of the following three modes (excluding starting pressure spikes and the meteoroid hazard):

1. Sublimation of the coating on the outside surface which decreases the outside surface emittance and results in excessively high wall temperatures. The high temperatures decrease the inside surface oxidation protection reliability and thus, the useful life.

2. During the transient time after engine shutdown, the combustion chamber pressure decreases rapidly relative to the wall temperature decrease. During the time at which the walls are relatively hot, a significant amount of the silicon is lost from the coating due to sublimation. Subsequent exposure to combustion gases degrades the coating. Repeated cycles of vacuum exposure and combustion gases results in a useable life less than would result from coating oxidation on the inside surface during continuous engine operation.

3. Reaction of the coating (initially $MoSi_2$) with the combustion gases (N_2, CO, H_2O, H_2, CO_2, O) changes the coating to compounds such as Mo_5, Si_3, and Mo_3Si, which has less oxidation resistance. The Mo_5Si_3 and Mo_3Si lose weight due to oxidation and erosion, with eventual coating wearout during continuous engine operation.

The combustor outside surface is grit blasted and then coated with Durak B, a proprietary $MoSi_2$ coating. The outside surface emittance at elevated temperature in vacuum is approximately the same for both grit blasted, non-coated molybdenum and disilicide coated molybdenum (Reference 3). In a hard vacuum, the outside surface coating is not required. If the outside surface coating completely sublimates, the combustor wall temperature should not greatly increase. The maximum combustor temperature is

2350°F. As a frame of reference, if the outside surface emittance decreased even to one-half of the value for an untested combustor, the maximum wall temperature would be equal or less than 2720°F. The outside surface coating is required for oxidation protection for engine testing at a cell pressure of 0.05 psia (Reference 2). Therefore, for engine operation in space, it is considered that coating sublimation on the outside surface will not have a significant effect on engine life.

The second mode of failure is a function of the duty cycle (the number of engine shutdowns for approximately steady state wall temperatures). The time for the combustor throat to cool down to 1000°F is of the order of one minute. Figure A-1 presents LMSC test data and calculated values of the disilicide coating weight loss versus time at 2600°F to 3200°F. The actual weight loss rate at a given temperature decreases from linearity with increasing time. The weight loss is initially governed by the linear rate of disproportionation of $MoSi_2$. Apparently, the diffusion of silicon to the surface subsequently becomes rate controlling. Lower silicides and molybdenum, Mo, formed on the surface effectively block the disproportionation of the underlying high silicides and diffusion of silicon through these outer layers becomes the weight loss rate control. The time at which non-linearity occurs increases with decreasing temperature. The calculated linear weight loss rates calculated by LMSC were based on the vapor pressure of $MoSi_2$, which was reported in Reference 8, and tabulated in Figure A-2. The same trend of decreasing rate of coating weight loss with time is shown in the tests reported by Hughes (see Figure A-3).

The empirical equation for the vapor pressure of $MoSi_2$ reported by Searcy and Tharp was used in extrapolating to lower temperatures, as shown in Figure A-4. The vapor pressure of $MoSi_2$ drops off rapidly at lower temperatures. The predicted maximum coating loss rate, based on the fastest evaporating compound $MoSi_2$, can be calculated using the Langmuir equation: (Reference 9).

$$R = \frac{p}{17.4} \sqrt{M/T} \qquad (1)$$

where:

R = rate of loss, grams/sec cm^2

p = vapor pressure, mm Hg at T

M = molecular weight of the material

T = material temperature, °K

WEIGHT LOSS VS TIME IN VACUUM

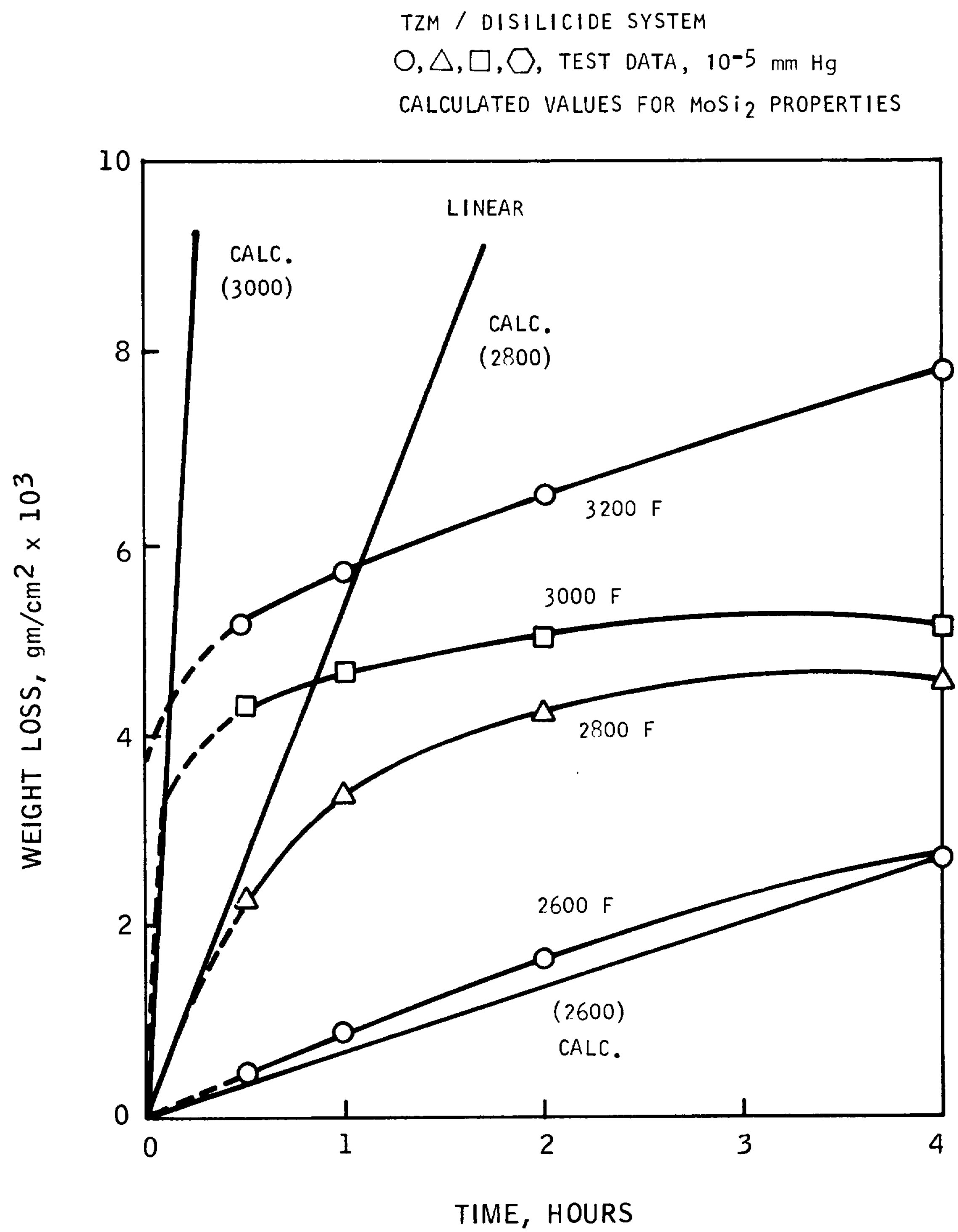


Figure A-1

CALCULATED VAPOR PRESSURE AND EVAPORATION RATE
OF Si FOR MOLYBDENUM SILICIDES

Temperature °F	Si Pressure mm Hg*			Rate of Si Loss $gm/cm^2/hr$		
	$MoSi_2$	Mo_5Si_3	Mo_3Si	$MoSi_2$	Mo_5Si_3	Mo_3Si
2600	2.49×10^{-5}	3.97×10^{-7}	2.68×10^{-7}	6.72×10^{-4}	1.07×10^{-5}	7.23×10^{-6}
2800	3.06×10^{-4}	4.26×10^{-6}	3.05×10^{-6}	5.38×10^{-3}	1.11×10^{-4}	7.98×10^{-5}
3000	1.30×10^{-3}	3.39×10^{-5}	2.57×10^{-5}	3.30×10^{-2}	8.61×10^{-4}	6.52×10^{-4}
3200	6.62×10^{-3}	2.12×10^{-4}	1.69×10^{-4}	1.61×10^{-1}	5.24×10^{-3}	4.17×10^{-3}

* Mo_3Si - $\text{Log } P = -33690/T - 5.75 \text{ Log } T + 28.94$

Mo_5Si_3 - $\text{Log } P = -32940/T - 5.75 \text{ Log } T + 28.67$

$MoSi_2$ - $\text{Log } P = -29800/T - 5.75 \text{ Log } T + 28.62$

Legend:

P - Atm

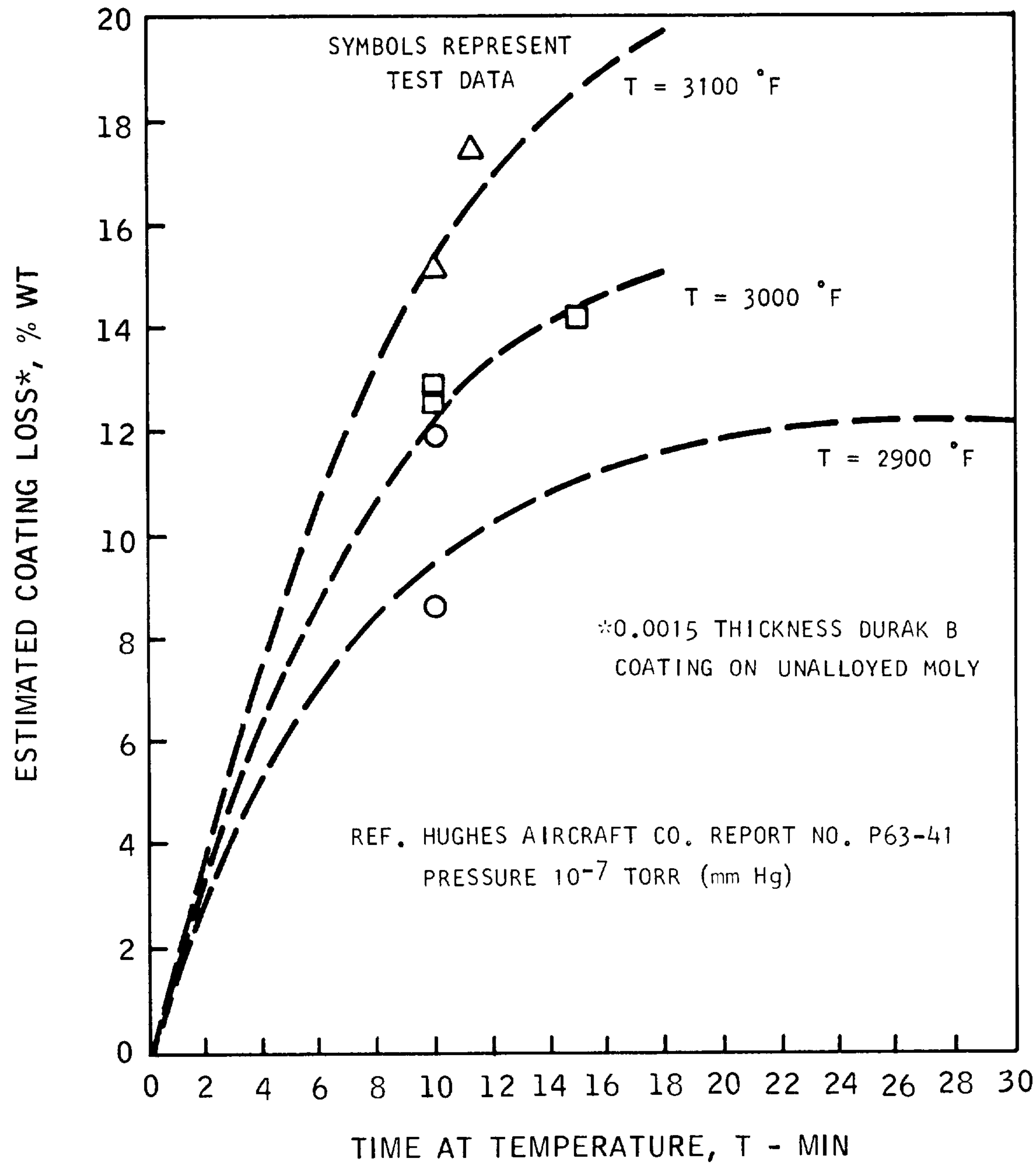
T - °K

Ref. 8 (Searcy and Tharp)

Calculated values of vapor pressure and rate of Si loss from Lockheed Report 2-04-64-2 (Reference 5)

Figure A-2

PREDICTED RATE OF DURAK B COATING LOSS WITH VACUUM EXPOSURE



VAPOR PRESSURE OF $MoSi_2$

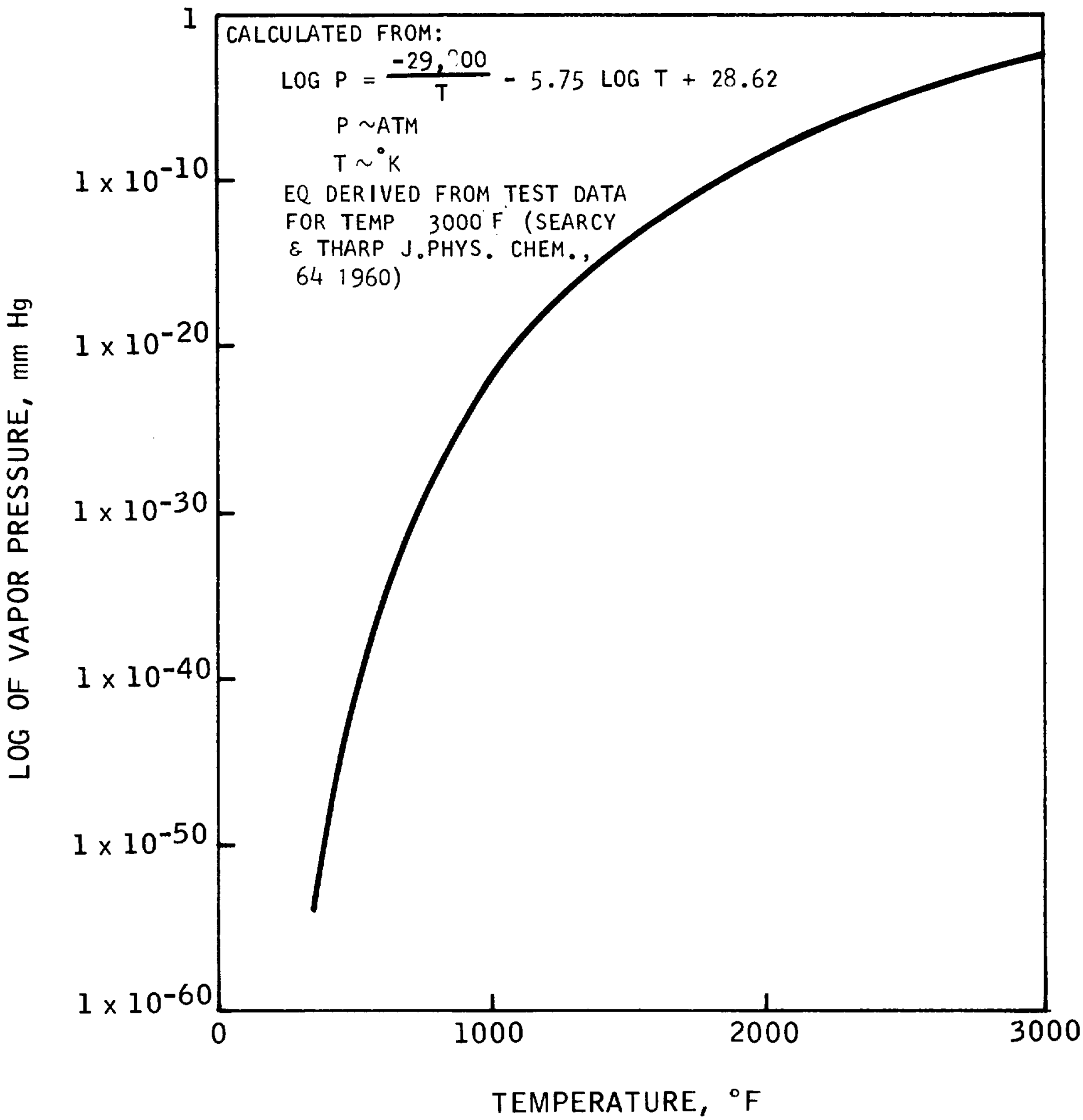


Figure A-4

The Langmuir equation is based on the assumption that all the atoms that escape from the surface are lost permanently. It yields the maximum rate of loss. Vapor pressure determined at ambient pressures of 10^{-5} mm Hg or lower, where the evaporating material strike cold walls and sticks, is not considered to differ significantly from space vacuum.

The maximum coating thickness loss, in inches/day, can be calculated from:

$$\dot{x} = \frac{R}{2.54 \rho} \qquad (2)$$

where:

$\dot{x}$ = rate of coating thickness loss, inches/day

ρ = density of material, gm/cm^3

2.54 = conversion factor, 1 inch = 2.54 cm

Using the vapor pressure of $MoSi_2$ from Figure A-4, a molecular weight of $MoSi_2$ of 152, and a density of 6.24, the maximum coating loss rate was calculated. See Figure A-5. The evaporation rate calculated should be a maximum, if the extrapolation of the vapor pressure to low temperatures is accurate, because the properties of the fastest evaporating compound $MoSi_2$ were used for the prediction. The coating loss at temperatures lower than 1500°F is practically negligible. During 600 days at 1500°F, the coating loss is predicted to be 0.33 x 10^{-8} inches. At 2000°F, the coating loss is predicted to be 0.0003 inch, or 15 percent, of the minimum initial coating thickness, during 600 days in a hard vacuum. Effectively no coating loss on the inside or outside surface of the combustor should occur for extended periods in the sun in space with the engine not operating.

Figure A-6 presents the predicted thrust chamber life versus the maximum wall temperature. The predicted life based on the extrapolation of the available engine test data is for essentially continuous engine operation at ambient pressures of approximately 0.05 psia. It is a prediction of the combustor life based on the third mode of failure which has been postulated. At a maximum combustor temperature of 2350°F, the combustor life should exceed one day of continuous engine operation in hard vacuum. Also shown in Figure A-6 is the predicted minimum time to sublimate an 0.002 inch thick disilicide coating in hard vacuum. The prediction should be most conservative, relatively, at 2800°F because of the coating loss rate decrease with time as shown in Figure A-1.

From the foregoing, it is apparent that no thrust chamber life prediction can be made for the postulated duty cycle oriented failure mode for extended Apollo missions. To determine if this actually does constitute a dominating failure mode, the following recommendations are made:

SUBLIMATION RATE OF DISILICIDE COATING IN HARD VACUUM

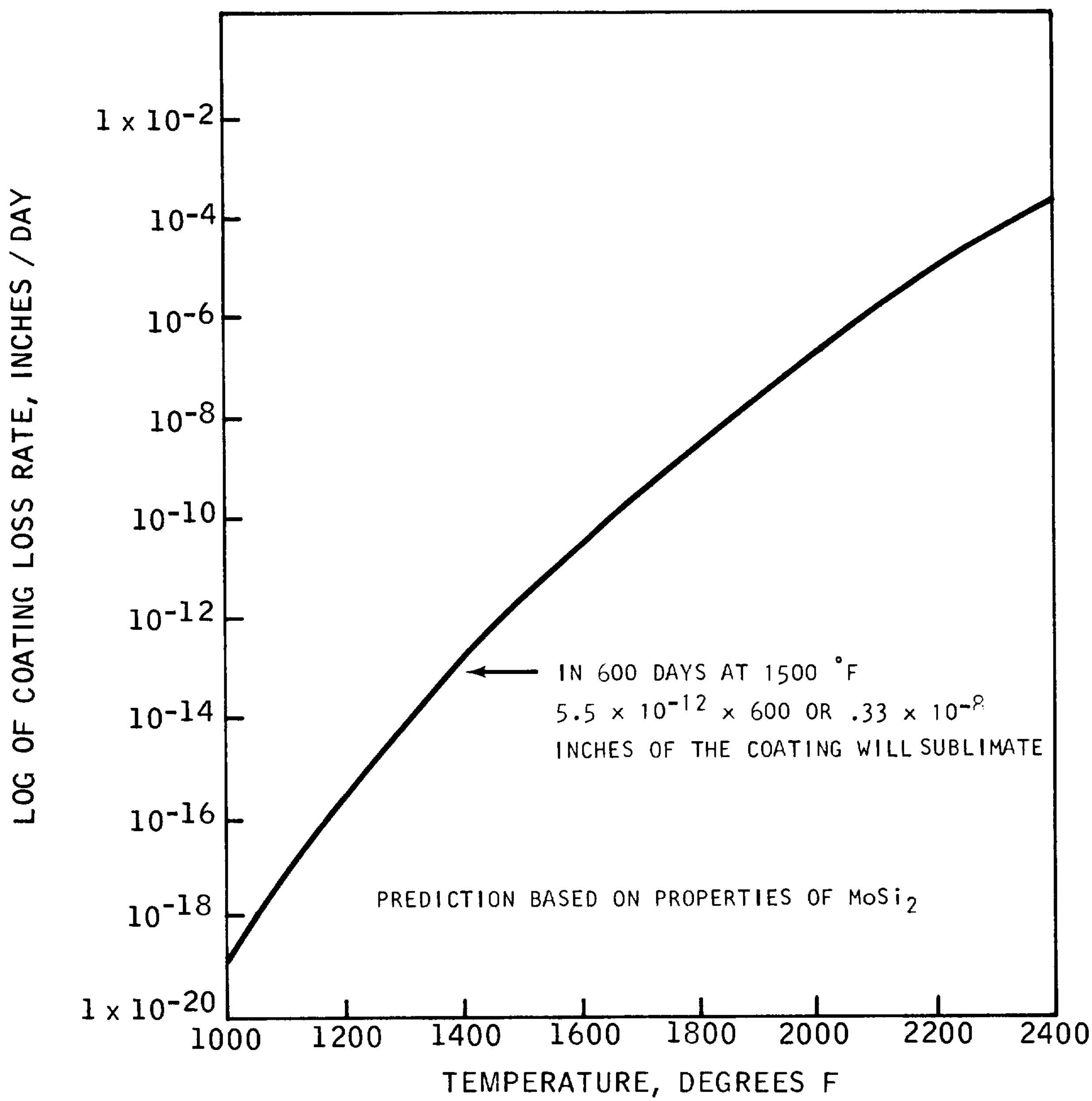


Figure A-5

PREDICTED COMBUSTOR LIFE

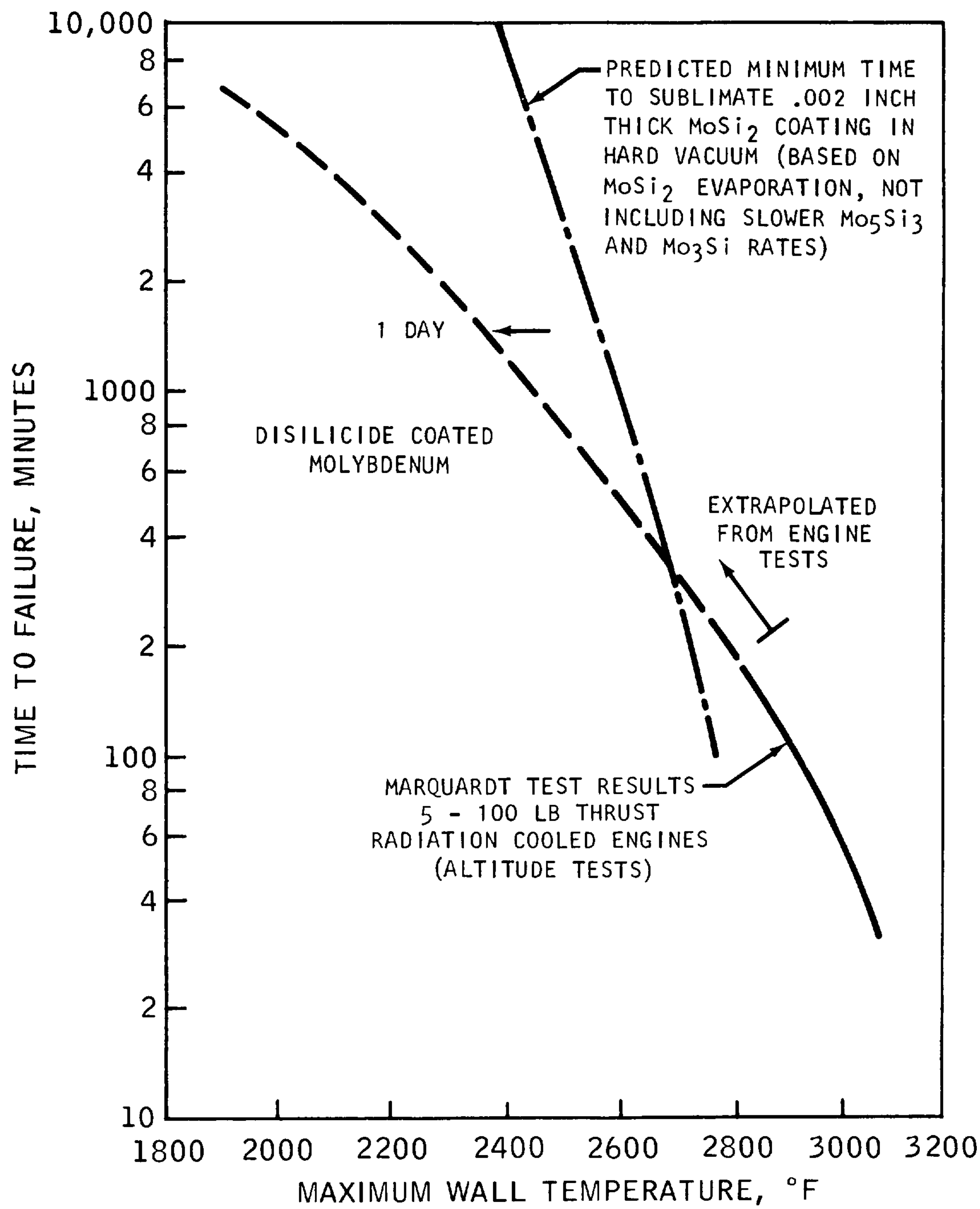


Figure A-6

1. Document combustor transient cool down temperatures.

2. Conduct laboratory tests on samples:

   a. Maintain sample at a representative transient cool down temperature (i.e., 2000°F) for several minutes at 10^{-5} mm Hg, corresponding to an accumulation of several cool down cycles.

   b. Subsequently, heat sample to 2350°F in an oxidizing atmosphere at (preferably) 95 psia.

   c. Repeat a and b above several times and determine coating degradation.

3. Conduct engine tests on the combustor:

   a. Maintain the combustor at a representative cool down temperature for several minutes at 10^{-5} mm Hg in a vacuum oven.

   b. Hot fire the engine at altitude.

   c. Repeat a and b above until engine failure occurs, or sufficient accumulated run time is achieved.

It is also recommended that analytical studies be conducted for the continuous operation failure mode and the duty cycle failure mode which have been postulated. This would involve estimating the chemical reactions and rates of reaction which occur during engine cool down and during engine operation. From the test results and studies, reliability predictions of combustor life should then be made.

REFERENCES

1. MPM 20.236A, "Project Apollo Phase I - Emittance Studies of Machine and Grit Blasted Molybdenum Tubes at 1-10 Microns Argon Pressure and 2900-3100°F Temperature", 24 February 1964.

2. MPM 20.237, "Project Apollo Phase II - Emittance Studies of Durak B Coated Smooth and Grit Blasted Molybdenum Tubes at 2580 μ Air Pressure (0.05 psia)", dated 5 February 1964.

3. MPM 17.174, "A Study of the Effect of Environment on Durak B Coated Molybdenum - One-half Percent Titanium Alloy and How Grit Blasting the Surface Affects Emittance", 12 March 1962.

4. Hughes Aircraft Company, Report No. P63-41, "Test Program to Study the Vaporization of Durak B in Vacuum at Elevated Temperature", August 1963.

5. Lockheed Missiles and Space Company, Report 2-04-64-2, "Coatings for Refractory Metals in Aerospace Environments", Contract No. AF 33(657)-11150, Fourth Progress Report, 1 March 1964 - 31 May 1964, 1 June 1964.

6. Lockheed Missiles and Space Company, Report 2-04-64-1, "Coatings for Refractory Metals in Aerospace Environments", Contract No. AF 33(657)-11150, Third Progress Report, 1 December 1963 - 29 February 1964, 2 April 1964.

7. Climax Molybdenum Company, Bulletin Cdb-6A, "Refractory Molybdenum Silicides", May 1963.

8. Searcy, A. W. and Tharp, A. G., "The Dissociation Pressures and the Heats of Formation of the Molybdenum Silicides", Journal of Physical Chemistry, Volume 64, October 1960, p. 1539-1542.

9. NASA TN D-477, "Survey of Materials Problems Resulting from Low Pressure and Radiation Environment in Space, November 1960, Acc. No. 32.160.

This page intentionally left blank.

APPENDIX B

METEOROID HAZARD TO THE SM/RCS ENGINES

INTRODUCTION

The meteoroid hazard, in terms of their number and mass, and the quantitative effect of their impact on specific materials and configurations is not well documented at the present time. (Reference 1). NASA, MSC Space Environment Division, has recommended the use of a specific meteoroid number-mass model and penetration equation from among the many existing models and equations to allow comparison of the predictions made by different organizations on the various space vehicle systems and components. The Whipple 1963 number-mass distribution corrected for a meteoroid density of 0.5 grams/cm^3 is used for both a 200 nautical mile earth orbit mission and a 100 nautical mile lunar orbit mission. (See Figure B-1). The recommended penetration equation is the Summer's single sheet penetration equation which is multiplied by a factor of 1.5 for application to thin sheet structures:

$$t = 1.67\,(m)^{1/3}\,(\rho_m)^{1/3}\,(V/\rho C)^{2/3} \tag{1}$$

where:

t = equivalent single sheet thickness of the structure, inches

m = mass of the meteoroid, grams

ρ_m = meteoroid density, gms/cm^3

ρ = density of the structural material, gms/cm^3

V/C = ratio of velocity of impact to speed of sound in the structural material

To determine the mass, m, of the meteoroid which is capable of penetrating the structural material, the structural material is converted to an equivalent thickness of aluminum:

$$t_{al} = t\left[\frac{(\rho E)}{(\rho E)_{al}}\right]^{1/3} \tag{2}$$

where:

E = modulus of elasticity

METEOROID FLUX - MASS DISTRIBUTION

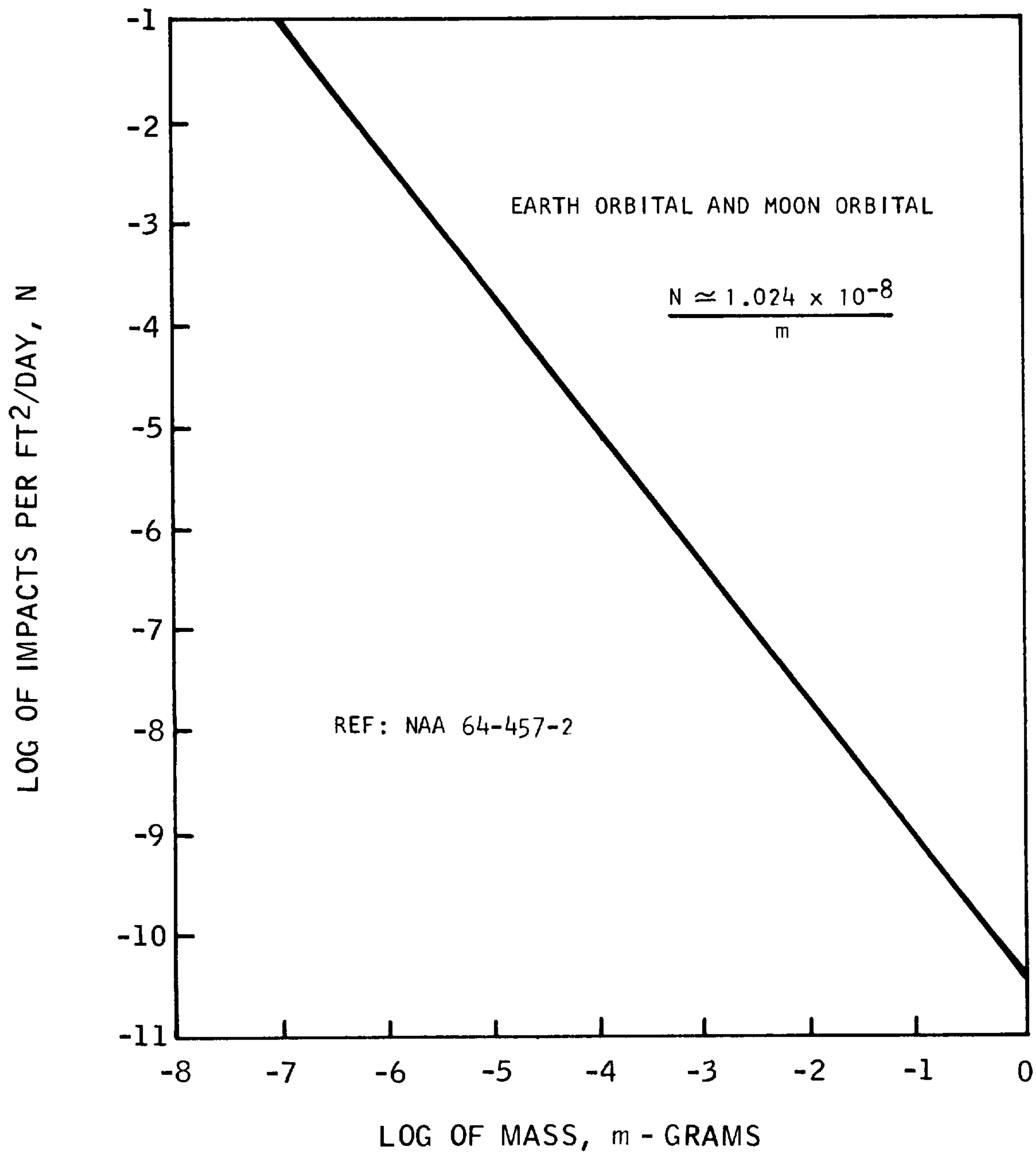


Figure B-1

The meteoroid mass which will just penetrate a structure can be calculated from equations (1) and (2). This calculated mass, m, determines the number, N, of meteoroid penetrations per ft^2 of surface per day. The effective number of penetrations, such as of the combustor interior coating, can then be estimated using shielding factors of the surface caused by the earth or moon, the vehicle, and the nozzle. The probability of no complete penetrations of the structure can then be estimated using the Poisson approximation of the binominal probability distribution:

$$P_o = e^{-F} \tag{3}$$

where:

P_o = probability of no penetration

F = total number of penetrations

The probability of no penetration of different sections of the engine can be estimated using the method outlined, which is the method used by North American Aviation, Inc., to predict the probability of no penetration of the command module structure.

The SM/RCS engines are installed on the spacecraft as shown in Figure B-2. The only part of the engine that is not protected from meteoroids by the shielding of the "doghouse" is the combustion chamber. The meteoroid hazard to the combustion chamber can be divided into three areas:

1. Penetration of the inside coating of the combustor by a meteoroid entering thru the exit nozzle.

2. Penetration of the combustor wall or spalling of the inside surface coating by meteoroid impact on the external surface.

3. Penetration of the L-605 expansion nozzles.

The analysis procedure utilized for areas (1) and (2) above are discussed in detail in the following sections. Analysis of area (3) requires only a straight-forward application of the procedure outlined above.

PENETRATION OF THE INSIDE COATING

The direct meteoroid hazard to the engine is primarily limited to penetration of the inside coating of the combustor, which is the combustion chamber and nozzle to an area ratio of 6.9. The probability of no complete disilicide coating penetration on the inside surface of the combustor is estimated below, based on References 1 and 2.

ENGINE MOUNT INSTALLATION

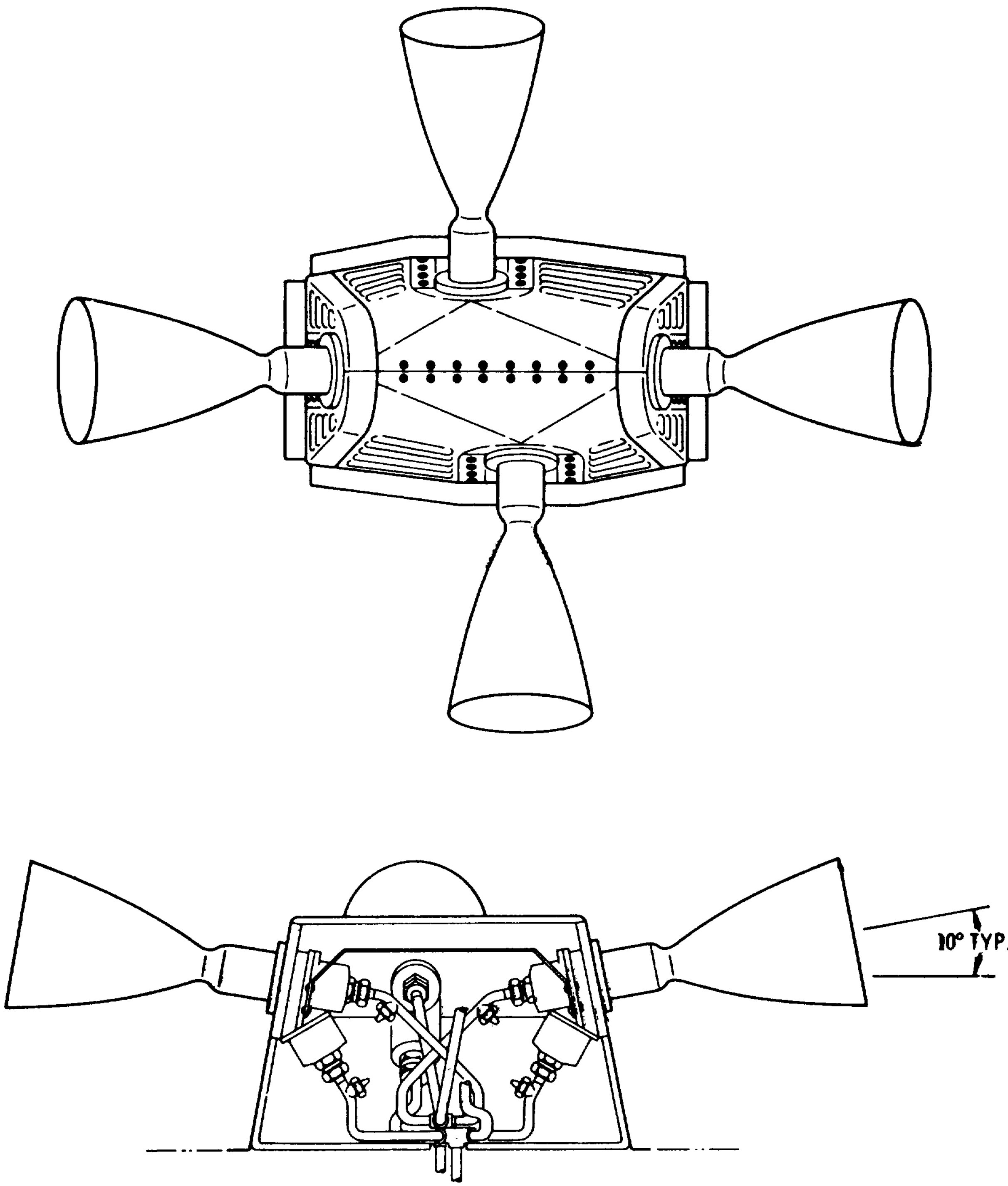


Figure B-2

For the disilicide coating (Reference 3):

ρ = 6.2 gms/cm^3

E = 59 x 10^6 psi

t = 0.0020 inch to 0.0032 inch

For aluminum:

ρ = 2.77 gms/cm^3

E = 10.5 x 10^6 psi

C = 5.104 kilometers/second

The meteoroid velocity, V, is assumed to be 30 kilometers/second. However, the inside surface is shielded by the nozzle, such that a meteoroid can impact on the surface at only an oblique angle. The maximum angle of impact is approximately 62 degrees. The maximum velocity normal to the coated surface is:

$$V_n = V \sin 62^\circ = 30 \times 0.883 = 26.5 \text{ km/sec}$$

From equation (2), the thickness of aluminum which is equivalent to the minimum disilicide coating thickness is:

$$t_{al} = 0.002 \left(\frac{6.2 \times 59 \times 10^6}{2.77 \times 10.5 \times 10^6} \right)^{1/3} = 0.00463 \text{ inch}$$

The coating thickness is small compared to the molybdenum wall thickness. The penetration thickness, from equation (1), which is for a single thin sheet, can be reduced by a factor of 1.5 for an infinitely thick plate. It is assumed, therefore, that since the coating is diffusion bonded to the base molybdenum, that the impact of low mass meteoroids on the coating is equivalent to meteoroid impact on a thick plate.

Substituting into equation (1):

$$0.00463 = \frac{1.67\ (m)^{1/3}}{1.5} (0.5)^{1/3} \left(\frac{26.5}{2.77 \times 5.104} \right)^{2/3}$$

Solving for m, the meteoroid mass just large enough to cause complete coating penetration:

$$m = 4.1 \times 10^{-8} \text{ gms}$$

From Figure B-1, N, the number of impacts per ft^2 per day is approximately 0.25.

The effective number of penetrations per day is:

$$\Psi = N \, \eta_{pl} \, \eta_{veh} \tag{4}$$

where:

Ψ = effective number of penetrations per day

η_{pl} = shielding effect of planet = 0.67 for a 200 mile earth orbit and 0.71 for a 100 mile lunar orbit

η_{veh} = shielding effect of vehicle on engine.

The vehicle shielding factor is analogous to a configuration factor as used in thermal or luminous radiation. The nozzle opening "sees" approximately 0.5 of half space, with the vehicle blocking off approximately 0.5 of the view.

Substituting into equation (4)

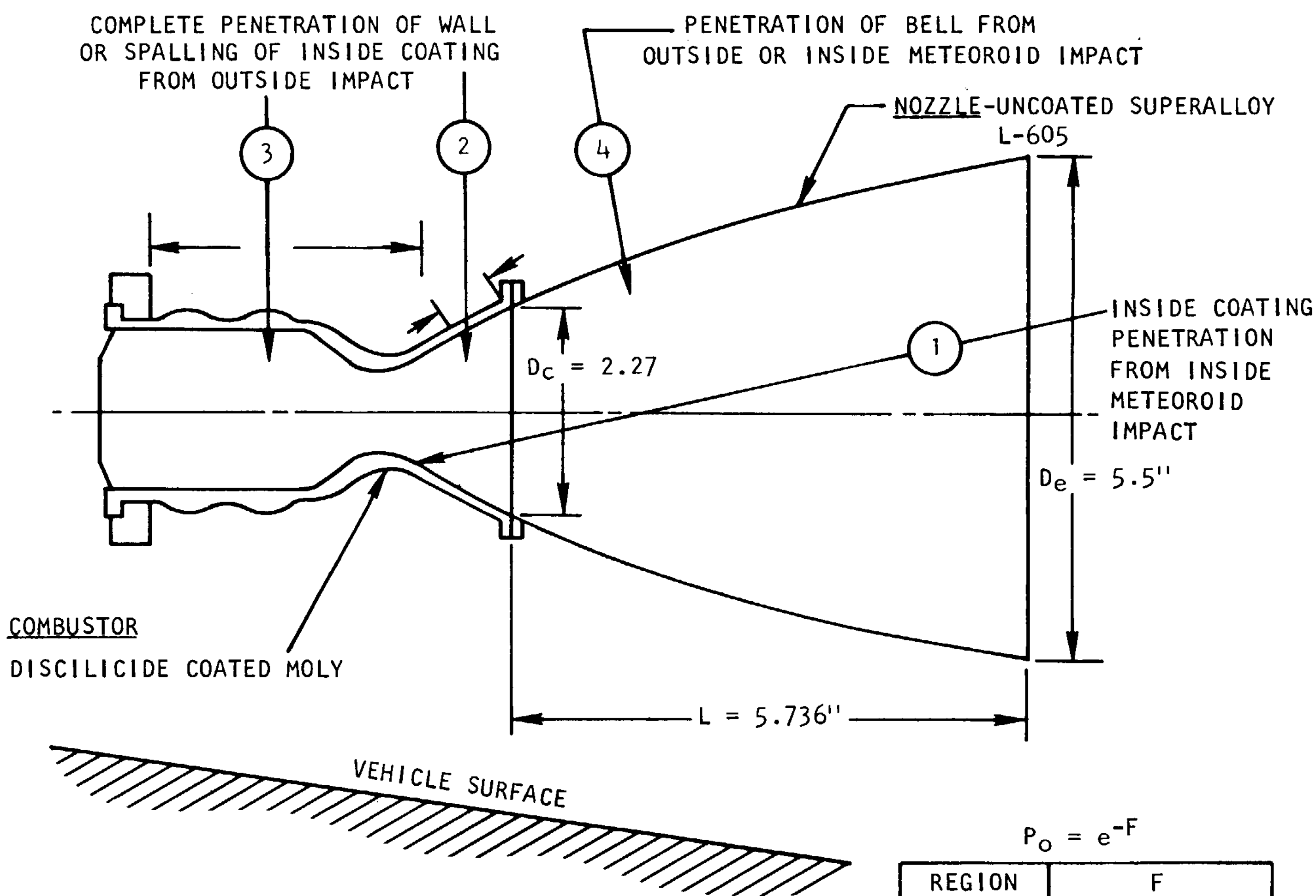
$$\Psi = 0.25 \times 0.71 \times 0.5 = 0.0887 \text{ impacts/ft}^2 \text{ day}$$

The effective surface area of the inside surface of the combustor is equal to the projected area of the end of the combustor times the configuration factor between this area and the exit area of the nozzle. Referring to the geometry shown in Figure B-3:

$$A_{eff} = \frac{\pi}{4} \times D_c^2 \, \phi \tag{5}$$

where ϕ for two parallel disks, from Reference 4 is:

ENGINE METEOROID HAZARD PREDICTION
APOLLO X MISSION



MISSION TIME, T (DAYS)	P_o = PROBABILITY OF NO COMPLETE PENETRATION OR SPALLING					
	1	2	3	4	2 OR 3	1 OR 2 OR 3
34	0.9851	0.99994	0.999982	0.9524	0.999925	0.9850
45	0.980	0.99992	0.999976	0.939	0.99990	0.9799

$P_o = e^{-F}$

REGION	F
1	4.37×10^{-4} T
2	1.68×10^{-6} T
3	5.28×10^{-7} T
4	1.40×10^{-3} T

Figure B-3

$$\phi = 1/2\left[1 + \frac{D_e^{\,2} + L^2}{D_c^{\,2}} - \sqrt{\left(1 + \frac{D_e^{\,2} + L^2}{D_c^{\,2}}\right)^2 - 4\left(\frac{D_e}{D_c}\right)^2}\,\right] \quad (6)$$

where

A_{eff} = effective surface area, ft^2

D_c = diameter of end of combustor, ft^2

D_e = diameter of end of nozzle, ft^2

L = length of L-605 nozzle, ft

ϕ = configuration factor

Substituting into equation (6):

$$\phi = 1/2\left[1 + \frac{(5.50)^2 + (5.736)^2}{(2.273)^2} - \sqrt{\left[1 + \frac{(5.50)^2 + (5.736)^2}{(2.273)^2}\right]^2 - 4\left(\frac{5.50}{2.273}\right)^2}\,\right]$$

$$\phi = 0.175$$

Substituting into equation (5):

$$A_{eff} = \frac{\pi}{4}\,\frac{(2.273)^2}{144} \times 0.175 = 0.492 \times 10^{-2}\ ft^2$$

This area is nearly entirely comprised of area downstream of the throat. The total number of penetrations, F, is:

$$F = \Psi\, A_{eff}\, T = 0.0887 \times 0.492 \times 10^{-2}\, T = 0.437 \times 10^{-3}\, T$$

From equation (3):

$$P_o = e^{-0.437 \times 10^{-3}\, T}$$

where T is the mission time in days.

The probability of no complete penetrations P_O, of the combustor inside coating versus mission time is shown in Figure B-4. For a 45-day mission, the probability of no complete penetration of the combustor inside coating is 0.98.

PENETRATION OF COMBUSTOR WALL OR SPALLING OF THE INSIDE SURFACE COATING

The combustor may also be damaged by spalling of the inside surface coating by meteoroid impact on the external surface which does not completely penetrate the combustor wall as well as the hazard of complete penetration. The most suitable approach, considering the current understanding of the problem of spalling related to thin walled structures, is to modify the Summer's single sheet penetration equation to account for spalling. (The equations and applicable nomenclature are presented in the Introduction above.) For a thin sheet the penetration thickness is multiplied by a severity factor of 1.5. To include the effects of spalling on the side opposite to the meteoroid impact, the severity factor should be increased to 2.0 to account for the combined effects of penetration and spalling. The penetration equation is then:

$$t = 2.23\ (m)^{1/3}\ (\rho\ m)^{1/3}\ (V/\rho\ c)^{2/3} \qquad (7)$$

The combustor can be divided into two regions:

1. An 0.045 in. (minimum) wall thickness region at the downstream end.

2. The combustion chamber region with an average wall thickness of 0.12 in. For the 0.45 inch thick region, the thickness of aluminum which is equivalent to the minimum wall thickness, from equation (2) is:

$$t_{al} = t\left[\frac{\rho\ E}{(\rho\ E)_{al}}\right]^{1/3} = 0.045\left(\frac{10.2 \times 46 \times 10^6}{2.77 \times 10.5 \times 10^6}\right)^{1/3} = 0.114 \text{ in.}$$

where ρ and E for molybdenum is 10.2 gms/cm^3 and 46 x 10^6 psi, respectively.

Substituting t_{al} into equation (7):

$$0.114 = 2.23\ (m)^{1/3}\ (0.5)^{1/3}\left(\frac{26.5}{2.77 \times 5.104}\right)^{2/3}$$

PROBABILITY OF NO COMPLETE PENETRATION OF THE INSIDE COATING OF THE COMBUSTOR VS MISSION TIME

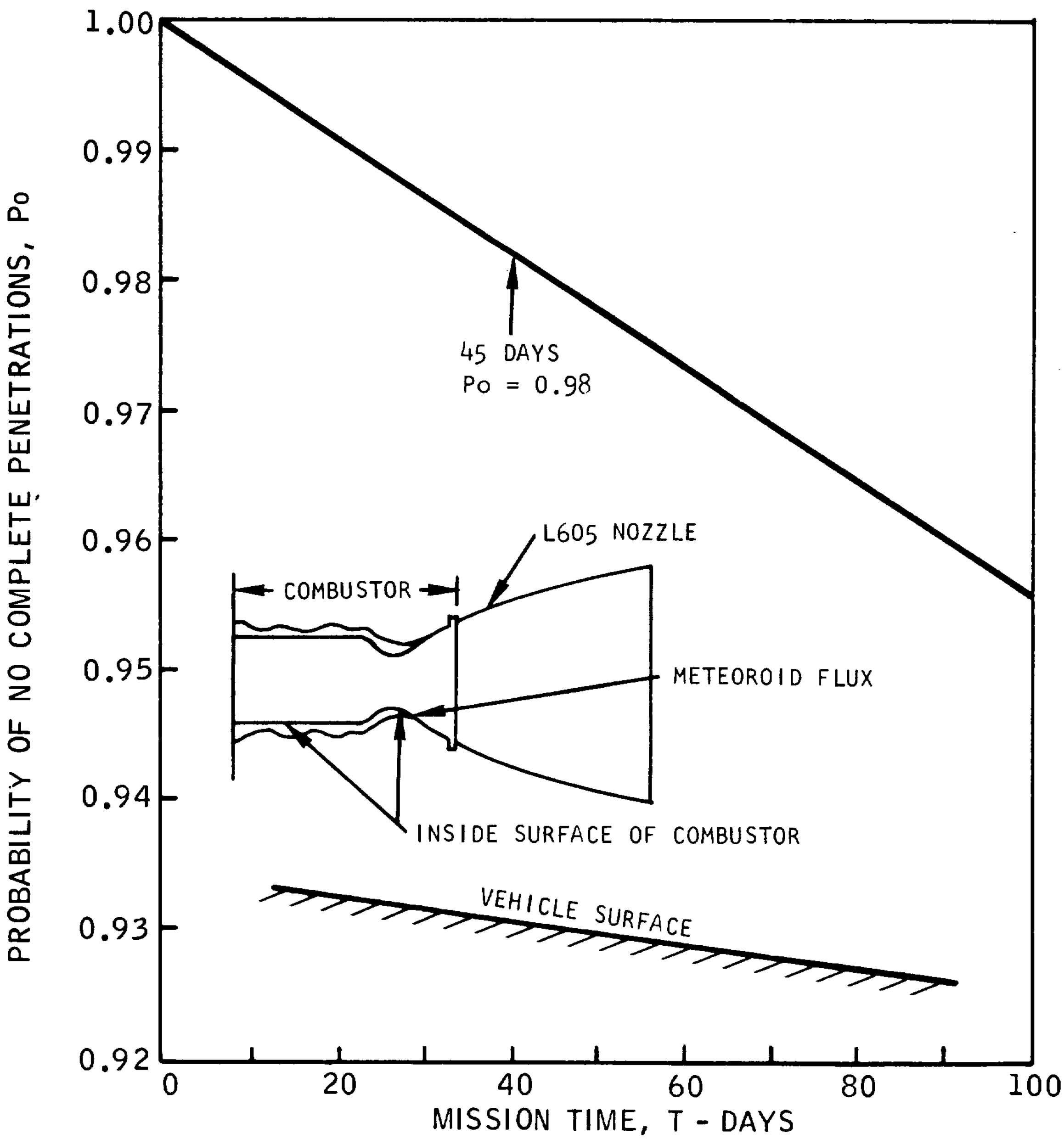


Figure B-4

Solving for m, the meteoroid mass just large enough to penetrate the combustor in the 0.045 inch thick region or to spall a section of the inside coating from an impact on the external surface:

$$m = 0.594 \times 10^{-4} \text{ gms}$$

From Figure B-1, the number N, of impacts per ft^2 per day for this meteoroid mass is approximately 1.73×10^{-4}. The surface area A_{eff} of the 0.045 inch thick region which includes an estimate of the shielding effect of the combustor nozzle attachment is approximately 0.0275 ft^2. The vehicle shielding factor for this entire surface area is approximately 0.5. The total number, F, of penetrations of the combustor or of meteoroid impacts causing spalling of the inside surface coating in the 0.045 inch thick region, from equation (4) is:

$$F = \Psi A_{eff} T = N \eta_{pl} \eta_{veh} A_{eff} T = 1.73 \times 10^{-4} \times 0.71 \times 0.5 \times 0.0275\, T = 1.68 \times 10^{-6}\, T$$

where T is the mission time in days and 0.71 is the planet shielding factor.

From equation (3), the probability of no damage of the inside surface coating from meteoroid impacts on the outside surface for the 0.045 inch thick region is:

$$P_o = e^{-1.68 \times 10^{-6}\, T}$$

Since the exponent is a small number, expansion of the exponential term results in:

$$P_o = e^{-F} \cong 1 - F \qquad (8)$$

The same procedure is used to evaluate the probability of no penetration or spalling of the inside coating of the combustion chamber (average thickness of 0.12 inch). The equivalent thickness of aluminum is 0.304 inch. The minimum mass, m, to damage the inside coating is 1.127×10^{-3} gms. The number of impacts per ft^2 per day is 0.91×10^{-5}. For a surface area of 0.164 ft^2, which includes an estimate of the shielding effect of the chamber injector attachment, a vehicle shielding factor of 0.5, a planet shielding factor of 0.71, the probability is:

$$P_o = e^{-5.28 \times 10^{-7}\, T} \cong 1 - 5.28 \times 10^{-7}\, T$$

The probability of either the combustion chamber (0.12 inch thick) or the downstream region of the combustion being damaged by external surface meteoroid impacts is equal to the summation of the probabilities of the individual occurrences. The probability of one meteoroid penetration or spalling of the inside surface for either region is:

$$P_F = 1 - P_o \qquad (9)$$

The probability of one meteoroid penetration or spalling in either of the two regions is therefore:

$$(P_F)_{\substack{\text{outside}\\\text{impact}}} = \left[1 - (P_o)_{0.045}\right] + \left[1 - (P_o)_{0.12}\right] \qquad (10)$$

The probability of no meteoroid damage to the inside coating from external meteoroid impact is:

$$P_{o\,\substack{\text{outside}\\\text{impact}}} = 1 - \sum P_F \qquad (11)$$

The individual probabilities and the combined probability of no penetration or spalling of the combustor inside surface coating from outside surface meteoroid impact is tabulated in Figure B-3.

CONCLUSIONS

The probability of the meteoroid hazard to the various areas of the combustion chamber is summarized in Figure B-3. If penetration of the L-605 nozzle does occur, the effect of a small hole or holes on engine performance should be negligible. Also penetration of the disilicide coating on the exterior of the combustion chamber should result in no significant increase in engine operating temperature due to the local loss of high emittance coating.

If penetration of the L-605 bell does not constitute a thrust chamber failure the dominating meteoroid hazard results from inside coating penetration from meteoroids passing through the exit plane of the nozzle. The probability of no combustor damage is essentially equal to the probability of no inside coating penetration from inside meteoroid impact.

REFERENCE

1. NAA Report SID 64-457-2 on Space Environment (Confidential)

2. IOM to G. Shein from C. S. Hill, "TMC 279: Combustion Chamber Meteoroid Hazard Evaluation," dated 8 October 1964.

3. TMC Report, "A Comprehensive Study of Anti-Oxidant Coatings for Commercially Pure Molybdenum," by C. Kallup and C. P. King, dated 1 September 1963.

4. Leuenberger and Person, "Compilation of Radiation Shape Factors for Cylindrical Assemblies," ASME Paper 65-A-144, ASME Annual Meeting, November 1956.

5. Personal communication B. Heckert (TMC) and G. Caylor (NAA), September 30, 1964; Subject: "Spalling Hazard".